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APOLLO COMMAND AND SERVICE MODULE SYSTEM SPECIFICATION (BLOCK 1) [U]

1 October 1964

NAS 9-150
Exhibit 1 Para. 4.2

Approved by

Dale D. Myers

Dale D. Myers
Vice President
Apollo Program Manager

~~at 3 year ... declassified after 12 years~~

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ILLUSTRATIONS

1. SCOPE

1.1 Scope.- This specification establishes the technical performance requirements for the development and design of the Apollo (Block I) Command and Service Module (CSM) System. The CSM System as discussed in this specification is comprised of a Launch Escape Subsystem (LES), a Command Module (CM), a Service Module (SM), a Spacecraft S-IVB Adapter, the associated Ground Support Equipment (GSE), and requisite Trainers. A general configuration of the Block I CSM and Launch Vehicles (LV) are delineated in Figures 1 and 2.

In addition, performance characteristics of the LV and the other items of Government-furnished equipment (GFE) upon which the design of the CSM System is based are specified.

1.2 Objective.- The objective of this specification is to serve as a contractual document to provide base line system requirements for the CSM and its supporting and associate systems.

2. APPLICABLE DOCUMENTS

The following documents form a part of this specification to the extent specified herein:

2.1 <u>Specifications</u>.- Documents identified with an asterisk (*) require future review and mutual agreement prior to incorporation into this specification.

2.1.1 <u>NASA</u>.-

MSFC-PROC-158A 12 April 1962	Soldering electrical connectors (high reliability) procedure for (amendments MSC-ASPO 5B of 18 May 1964
MSFC 10M 01071 6 March 1961	Environment-protection When Using Electrical Equipment-Within the Areas of Saturn Complexes Where Hazardous Area Exist; procedure for
	Exhibits for the Apollo Block I Space Suit Assembly Procurement Package, estimated date October 20, 1964.
MSC-GSE-1B 12 February 1964	Apollo Ground Support Equipment General Environmental Criteria and Test Specification

2.1.2 <u>Military</u>.-

MIL-E-6051C 17 June 1960	Electrical Electronic System Compatibility and Interference Control Requirements for Aeronautical Weapon Systems, Associated Subsystems and Aircraft.

2.1.3 <u>Contractor</u>.-
North American Aviation, Inc. Space and Information Systems Division (NAA/S&ID)

MC 999-0002B 3 January 1963	Electromagnetic Interface Control for the Apollo Space System
* SID 62-109, Vol. 5	Apollo Spacecraft Development Test Plan

* SID 62-203 27 December 1963	NAA/S&ID Reliability Program Plan
* SID 62-1000 17 September 1964	Preliminary Guidance and Navigation System Performance and Interface (P&I) Requirements Specifications
* SID 62-1001 17 September 1964	Flight Research and Development (R&D) Instrumentation Specification
* SID 62-1002 17 September 1964	Scientific Equipment Interface Specification
* SID 62-1003 17 September 1964	Preliminary NASA-Furnished Crew Equipment Interface and Performance Specification
* SID 63-489	Master Ground Operation Spacecraft
* SID 64-1237 4 September 1964	Vehicle Model Specification, Basic - Block I
* SID 64-1807	Model Specification, Apollo Training Equipment
* SID 64-1866	CSM/MSFN Communications Interface Specification

2.2 Standards.-

2.2.1 Federal.-

ARDC 1959 Atmosphere
Standard Atmosphere 1962

Federal Standard 209 16 December 1963	Clean Room and Work Station Requirements, Controlled Environments

2.2.2 Military.-

MIL-STD-130B 24 September 1962	Identification Marking of U.S. Military Property
MS 33586A 16 December 1958	Metals; definitions of dissimilar

2.2.3 Contractor.- Not applicable

2.3 Drawings.-

ICD 13M20108 20 July 1964	"Instrument Unit to Spacecraft Physical Requirements" (Saturn lB)
ICD 13M20109 24 July 1964	"Spacecraft/"Q"-Ball Physical Requirements" (Saturn lB)
ICD 13M50103 28 May 1964	"Instrument Unit to Spacecraft Physical Requirements" (Saturn V)
ICD 13M50112 22 July 1964	Spacecraft/"Q"-Ball Physical Requirements (Saturn V)
ICD 13M50123 29 July 1964	Envelope LEM/S-IVB/IU Clearance, Physical

2.4 Bulletins.- Not applicable

2.5 Other Publications.-

2.5.1 NASA.-

NPC 200-2 20 April 1962	Quality-Program Provisions for Space System Contracts
NPC 200-3 20 April 1962	Inspection System Provisions, Suppliers of Space Material, Parts, Components, and Services
NPC 250-1 July 1963	Reliability Program Provisions for Space Systems Contractors

2.5.2 Military.-

AFETC Pamphlet 80-2, Vol. I 1 October 1963	General Range Safety Plan, Prelaunch Safety Procedure
AFCRL 62-899 July 1962	Two Point Variability of Winds, Vols. I, II and III.
WADC TR 52-321 September 1954	Anthropometry of Flying Personnel - 1950

2.6 <u>Precedence</u>.- The order of precedence in case of conflict shall be as follows:

a. The Contract, NAS9-150,

b. This specification

c. SID 64-1237, Vehicle Model Specification -

d. Other documents referenced herein

2.7 <u>Effectivity</u>.- Effectivity of contract changes shall be as of the date of this document.

3. REQUIREMENTS

The basis for design of the CSM System shall be a lunar orbit mission which may be defined as the lunar orbit rendezvous (LOR) mission without the lunar excursion module (LEM) interface and with certain other deviations as specified in this document.

This section will encompass the following:

a. Definition of major elements of the system.

b. Design constraints and standards necessary to assure compatibility of program hardware.

c. The allocation of performance budgets and specified design constraints.

d. Identification of principal functional interfaces.

e. Identification and use of the Government-furnished equipment (GFE) forming an integral part of the system.

3.1 General Requirements.- These general requirements are the collection of principles to which the basic technical approach of the CSM subsystems must be responsive. They are the first order criteria from which successive design criteria, performance margins, tolerances, and environments shall be developed.

3.1.1 Command and Service Module.

3.1.1.1 Command Module (CM).- The CSM shall include a recoverable CM. This module shall contain the communication, navigation, guidance, control, computing, display equipment, and other equipment requiring crew mode selection. In addition, other equipment required during nominal or emergency Earth landing phases shall be included in the CM. This module shall include features which allow effective crew observation with a field of view for general observation. Equipment arrangements shall allow access for maintenance prior to Earth launch. The CM shall provide for sufficient storage of experimental measurement equipment as specified in SID 62-1001, Flight R&D Instrumentation Specification.

a. Housing - The CM shall house three crew members during the launch, translunar, lunar orbit, transearth entry and recovery phases.

b. Entry and Earth Landing - The CM shall be the entry and Earth landing vehicle for both nominal and emergency mission phases.

c. Ingress and Egress - The side ingress and egress hatch to the CM shall be used during countdown or recovery. No provisions shall be made for extra-vehicular activity during flight.

3.1.1.2 Service Module (SM).- An unmanned SM will be provided for all missions. This unmanned module shall contain stores and systems which do not require crew maintenance or direct operation, and which are not required by the CM after separation from the SM. The SM shall house all propulsion subsystems required for midcourse corrections, lunar orbit insertion, lunar orbit maneuvers and transearth injection. The SM will be jettisoned prior to entry into the Earth's atmosphere.

3.1.1.3 Command and Service Module SIV-B Adapter.- The CSM/S-IVB Adapter shall structurally and functionally adapt the SM to the LV, and provide for in-flight separation of CSM from the LV.

3.1.1.4 Launch Escape Subsystem (LES).- Provisions shall be made to separate the CM from the LV in the event of failure or imminent failure of the LV during all atmospheric phases.

3.1.1.5 Command and Service Module Subsystems.- The CSM subsystems requirements and subsystem descriptions are contained in SID 64-1237, Vehicle Model Specification - Basic.

3.1.2 Operational Concept.

3.1.2.1 Manning of Flight.- The CSM shall be designed for manned operation with full utilization of human crew capabilities. Automatic subsystems shall be employed only where they will enhance the performance of the mission. Where possible, automatic systems shall be manually operable by override.

3.1.2.2 Onboard Command.- The spacecraft will normally utilize inputs from earth-based tracking and computing facilities in conjunction with onboard computations to perform mission requirements. However, the spacecraft shall have the capability of performing any phase of the mission independent of ground facilities or of aborting the mission in its entirety.

3.1.2.3 Flight Crew.- The CSM flight crew shall consist of three men.

3.1.2.3.1 Crew Participation.- The flight crew shall have the capability to control the CSM throughout all flight modes. The flight crew shall participate in navigation, control, monitoring, computing, and observation as required. Status of subsystems shall be displayed for crew monitoring, failure detection and operational mode selection. The CSM shall be designed so that a single crewman will be able to perform all tasks essential to return the CSM in case of emergency.

3.1.2.3.2 Abort Initiation.- Provisions shall be made for crew initiation of all abort modes. Initiation of abort modes by automatic subsystems shall be provided only when necessary to insure crew safety.

3.1.2.4 Flight Time Capabilities.

3.1.2.4.1 Flight.- The Apollo CSM shall be designed to accomplish the Lunar Orbit Mission. The CSM consumables subsystems shall be designed for a nominal mission time of 10.6 days with 3 of these days in lunar orbit using the Δ V allocations shown in 3.1.3.3. By judicious system management of duty cycles, alternate missions, such as Earth orbital, may be performed within the resultant capabilities of the CSM system. A 14-day mission capability is inherent by proper selection of duty cycles for use of the total electric power available since the electrical power cryogenics was sized to permit abort from the most critical point in the lunar mission with one cyro tank inoperative. In addition, provisions are available for on-landing of additional lithium hydroxide canisters, and the oxygen supply is sized to provide CM leakage of 0.2 pounds per hour plus metabolic oxygen, as required for a 14-day mission.

3.1.2.4.2 Post Flight.- The CM shall provide the crew with a habitable environment for 48 hours and a flotation environment of 7 days following a water landing.

3.1.2.5 Earth Landing.- The CSM shall have the capability of initiating a return and earth-landing maneuver at any time during either lunar or Earth orbital missions. Prior to each flight, a primary water landing site and suitable backup water landing site will be selected for normal mission landing. Emergency land landing capability shall be provided for early launch aborts.

3.1.3 Command and Service Module Performance.- The following subparagraphs summarize the nominal performance capabilities of the CM, SM, and S-IVB Adapter.

3.1.3.1 Boost Stabilization.- The effects of windage, aerodynamics, variations of the center of gravity, etc., will be compensated for by the launch vehicle during the boost phase.

3.1.3.2 Trajectories.- The general CSM trajectories shall follow the general requirements described in 3.1. After translunar injection, the primary measured CSM positional accuracy shall be provided by the MSFN with the CSM G&N Subsystem serving as a back-up system in accordance with SID 64-881, NASA Manned Space Flight Net Performance and Interface Specification - Primary.

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3.1.3.3 Command and Service Module Propulsion Increments After SIVB Separation.- After separation of the SIVB, propulsion increments of the CSM shall be supplied by the SPS. For mission comparison purposes, weight repo etc., the SPS characteristic velocity budget utilized shall be as indicated for the following mission phases.

Mission Phase	Incremental Velocity (FPS)
a. Translunar	
(1) Midcourse	300
(2) Lunar orbit injection	3,230
(3) Δ V margin (10 percent)	353
b. Transearth	
(1) Lunar orbit maneuvers	455
(2) Transearth injection	3,610
(3) Transearth midcourse	300
(4) Δ V margin (10 percent)	436

3.1.4 Mission Performance.

3.1.4.1 Flight Plan.- The Apollo mission flight plan for which the CSM is sized shall be as specified in 3.1.4.1.1 and 3.1.4.1.2.

3.1.4.1.1 General Flight Plan Requirements and Characteristics.- The general flight plan requirements and characteristics present the general mission ground rules to which the CSM shall be designed. These ground rules consist of trajectory parameters and operational constraints which shall be used in overall CSM and subsystem design. The characteristics of the lunar missions are described in the following subparagraphs:

a. Launch site - All lunar orbital missions shall be launched from Cape Kennedy, Florida. The launch azimuth shall be within limitations set by range safety and tracking considerations. The launch phase for lunar orbital missions begins within S-IC ignition and ends with S-IVB cutoff in Earth parking orbit.

b. Launch time window - Lunar orbital mission flight plans shall include at least a 2-hour period on the launch date. Launches may be made providing visual reference conditions sufficient for orientation during high altitude abort exist. A launch window shall be provided either by maneuvering the CSM space vehicle to intercept a planned trajectory or by selecting a new trajectory that will satisfy the mission objectives and which will also be obtained at the actual launch time. Both the lunar trajectory selection and vehicle maneuvering methods shall be developed for obtaining a launch window. This capability is to be provided by a Government-furnished launch vehicle from a Government-furnished launch complex.

c. Earth parking orbit - The Earth parking orbit phase begins with S-IVB cutoff in orbit and ends with S-IVB relight for translunar injection. The parking orbit altitudes for lunar orbital mission shall be limited to altitudes from 90 to 120 nautical miles. The nominal parking orbit altitude shall be 100 nautical miles. Multiple parking orbits are acceptable but shall be compatible with booster performance and lifetime limitations. The duration of this phase shall not exceed 4 1/2 hours.

d. Translunar injection - The translunar injection phase begins with S-IVB ignition in Earth parking orbit and ends with S-IVB cutoff. Final injection into the translunar trajectory shall be located such that the trajectory can be determined by the MSFN within 15 minutes of translunar injection burnout.

e. Translunar coast - The translunar coast phase begins with S-IVB cutoff and ends with SPS ignition for lunar orbit insertion. The translunar trajectory for lunar orbit missions shall be a free return type which has a coast return to the Earth with acceptable entry conditions. The duration of this phase shall be from approximately 59 to 77 hours depending upon the Earth-Moon distance, the inclinations of the geocentric translunar and transearth planes to the Moon's orbit plane, and the injection velocity. The translunar trajectories for lunar orbit missions shall have a nominal pericynthion of 100 nautical miles. The CSM shall include provisions for performing translunar midcourse correction maneuvers.

f. Lunar orbit insertion - Lunar orbit insertion begins with SPS ignition just prior to pericynthion and ends with SPS cutoff in lunar orbit. Insertion will occur over the nonvisible portion of the Moon. The CSM shall arrive on a circumlunar trajectory which has a nominal pericynthion altitude of 100 nautical miles. A 5-degree plane change capability shall be provided within the lunar orbit injection velocity budget for establishing the initial orbit. The maneuver shall be accomplished at the same time as the retro-maneuver for establishing the lunar orbit.

g. Lunar orbit - The lunar orbit phase begins with SPS cutoff in lunar orbit and ends with SPS ignition for transearth injection. The nominal lunar orbit altitude shall be 100 nautical miles.

h. Transearth injection - Transearth injection begins with SPS ignition in lunar orbit and ends with SPS cutoff. The SM propulsion subsystem shall be capable of providing the necessary propulsion performance to transfer from the lunar orbit to the transearth trajectory. The maneuver required is a function of the characteristic of parking orbit at the time of injection, the time spent in orbit, and the terminal constraints at perigee which must be satisfied. The terminal constraints which must be satisfied are the Earth atmospheric entry angle, geocentric conic inclination, and the entry epoch. The required entry angle shall be limited such that capture is insured without exceeding the aerodynamic heating or loads limitations. The position of the vehicle at the time of injection will be over the non-visible side of the Moon.

i. Transearth coast - The transearth coast phase begins with SPS cutoff and concludes at the entry interface. The duration is determined by the transearth injection conditions and shall range between 60 and 84 hours to allow for return to the primary landing site. The inclination of the transearth trajectory to the Earth's equator and the time of flight shall be used to control the entry in such a way that the entry track will be over planned tracking and recovery areas. The CSM shall include provisions for performing transearth midcourse correction maneuver. Transearth trajectories shall be such that nominal entry for Apollo missions will be with posigrade motion with respect to the Earth to reduce the entry heating and widen the entry corridor.

j. Entry - The entry phase begins at the entry interface (nominally 400,000 feet) and ends at drogue parachute deployment. The CM shall be capable of entry with a normal operational corridor with a maximum deceleration of 9 g during the initial pull-out with a minimum L/D=0.30 and at a parabolic velocity of 36,333 fps when measured in a vacuum at perigee. The maximum range at minimum L/D shall be 2,500 nautical miles. The maximum emergency deceleration limit shall not exceed 20 g.

k. Recovery - The recovery phase covers the time commencing with drogue parachute deployment and ending with touchdown of the CM.

l. Post landing - The post recovery landing covers the time from CM touchdown to CM retrieval.

3.1.4.1.2 Control Weight and Consumables Design Mission.- The following 10.6-day lunar orbit mission timeline shall serve as a basis for provisioning of consumables and for establishing CSM control weights:

Mission Phase	Duration Hours
Prelaunch	10.00
Ascent phase	0.19
Earth parking orbit	4.40
Translunar injection	0.09
Translunar coast	77.00
Lunar orbit injection	0.09
Lunar orbit coast	88.00
Transearth injection	0.04
Transearth coast	84.00
Pre-entry	0.08
Entry	0.50
Recovery	0.17

3.1.4.2 Contingencies.- A contingency situation is the result of any deviation from the mission plan which requires a decision to be made concerning future conduct of the mission. Such deviations can include those concerned with schedule, structural characteristics, vehicle or subsystem performance, crew condition, random natural hazards and others. The CSM individually, and the overall Apollo system, as a whole, shall be capable of resolving contingencies in order to meet the specified probabilities of crew safety and of mission success. The spacecraft shall be designed such that any one crewman can perform all functions required to accomplish a safe return to earth from any point in the mission.

3.1.4.2.1 <u>Design Objective</u>.- Overcoming contingency situations requires operational and performance flexibility. This flexibility shall be provided by the following design objectives:

a. Built-in redundancy

b. Switch-in redundancy

c. Alternate operating modes

3.1.4.2.2 <u>Criteria for Contingency Operation</u>.- Performance requirements for CSM operation under contingency conditions shall be based on the following criteria (listed in approximate order of significance):

a. Adequate crew safety

b. Mission success

c. Adequate fuel margin

d. Minimum response-time criticality

e. Primary landing area

f. Adequate margin for consumables

g. Manned Spacecraft Control Center (MSCC) and MSFN assistance

h. Hardware reliability

i. Minimum number of abort trajectories

j. Minimum flight-plan complexities

k. Performance flexibility

3.1.4.2.3 <u>Contingency Operations</u>.- Crew response to a contingency will comprise, in general, the operations described below.

a. Detection of contingency - The crew members shall be alerted to the contingency occurrence by one or more of the following:

   (1) CSM displays and controls

   (2) Telemetry/communication loops

   (3) Telemetry/up-data link

(4) Lack of response to command inputs

(5) Physical sensing by astronaut

(6) Caution and warning display

b. Isolation of contingency - To aid the crew in isolation of contingencies, all information required to assure crew safety shall be stored on board the CSM in a readily accessible manner. Pertinent information affecting mission success shall be stored on board where practicable. Complete information at all levels and quantitative predictions of future missions status shall be available from MSCC via MSFN within existing communications capabilities.

c. Evaluation of contingency - On-board stored contingency data shall clearly identify contingencies where crew safety may be jeopardized and where time may be a constraining factor.

d. Implementation of contingency resolution - The resolution of all contingencies shall be initiated by the crew. Automatic initiation shall be invoked only when the response time or the complexity of the evaluation and implementation process are beyond reasonable human limitations.

3.1.4.2.4 Abort Factors.- For abort action, the on-board stored contingency data shall normally provide abort-selection criteria including propulsive fuel, time, and landing area.

a. Propellants - Data listing ΔV requirements for discrete abort trajectories shall be readily available on board. Sufficient conversion data shall be available on board to convert propellant readings to ΔV capabilities.

b. Time - Time histories for discrete abort trajectories shall be readily available on board. Sufficient information concerning consumable usage rates under varying operational conditions shall also be available on board to enable reasonable predictions on future consumable status. In addition, those contingencies which require a timely response shall be identified in the on-board stored data.

c. Information retrieval - On-board stored data shall be in sufficient detail to provide adequate assurance of crew safety even if communications to MSCC are not available. An efficient unambiguous indexing method shall be provided to enable speedy retrieval by the astronauts of adequate information from the on-board stored data.

3.1.5 Launch Vehicle (LV) Performance Requirements.- Propulsion increments involved with the boost phases of the mission will be supplied by NASA-furnished Saturn 1B or Saturn V launch vehicles. The CSM system shall be designed compatible with the following interface requirements.

3.1.5.1 Launch Vehicle Attitude Control.- The limit cycle or dead band for the attitude control subsystem of the LV shall be $\pm$ 1.0 degree in pitch, roll and yaw at a rate not to exceed 0.05 degrees per second.

3.1.5.2 Propellant Venting.- The SIVB propellant venting shall be continuous and the thrust generated shall not cause any moment that cannot be corrected within the attitude control subsystem dead band.

3.1.5.3 Loads Criteria for CSM SIVB Adapter and Instrumentation Unit (IU).- The following maximum flight parameters shall not be exceeded on Block I missions.

3.1.5.3.1 Load Parameters

$$\alpha = 6.9 \text{ degrees}$$

$$q\,\alpha = 5{,}072 \text{ lb/ft}^2 \text{ deg. (for flexible body conditions)}$$

Trajectories shall be shaped such that these parameters are not exceeded. (see Figure 3)

The limit axial loads, shears, and bending moments at the CSM/LV interface shall not exceed those shown in the following table for conditions of maximum $q\,\alpha$:

Load Parameters

Loads x 10^{-3}

Interface	S Shear (lbs.)	P Axial Load (lbs.)	Moment M_A (lbs.)	M Moment ΔM_z (in-lb)	Moment $\angle M_y$ (in-lb)
Adapter/IU (Apollo Station 502)	63.0	-269.0	21,800	260	6.3

Note: MA is the moment due to the trajectory and trajectory dispersions. ΔM_z and ΔM_y are fixed direction moments are due to the physical design of the vehicle (CG locations, asymmetry, etc.)

3.1.5.3.2 Booster Aero Data.- Loads evaluated for the max $q\alpha$ condition shall use the Booster normal force coefficient and center of pressure as given in Figure 4, and the normal force distribution, for the booster only, as given in Figure 5.

3.1.5.3.3 Booster Control System Criteria.- The system gains for the autopilot are derived by using a combination of the minimum drift and minimum load principles having an auxiliary feedback loop utilizing an angle of attack or an accelerometer sensor.

3.1.5.3.4 Booster Stiffness.- Distributions of EI and KAG for the Saturn V booster are given in Figures 6 thru 9.

3.1.5.3.5 Booster Weight Distribution.- Weight distributions for the Saturn V booster structure and propellant are given in Figures 10 thru 14.

3.1.5.4 Saturn IB Performance Requirements:

3.1.5.4.1 Payload Capability.- For the Saturn IB missions the LV shall be capable of injecting 32,500 pounds, based on a 8,200-pound Launch Escape Subsystem (LES), into a nominal 100-nautical mile Earth orbit.

3.1.5.4.2 Trajectory Requirements, Saturn IB.- The LV shall insert the CSM at cutoff into the particular mission design trajectory within the accuracies defined by the following flight parameters:

Eccentricity, e,	± 0.00725
Semi major axis, a.	± 10.70 nautical miles
Ascending node,	± 0.210 degrees
Inclination,	± 0.0839 degrees
True anomaly of the ascending node (If applicable for the particular mission)	± 0.830 degrees

3.1.5.5 Saturn V Performance Requirements.-

3.1.5.5.1 Payload Capability.- For the Saturn V missions the LV shall be capable of injecting 90,000 pounds, based on an 8,200-pound LES and a 24,500-pound LEM into a translunar trajectory of the free return type having a nominal vacuum perigee altitude of 21 nautical miles with no midcourse corrections required to accomplish the trajectory.

3.1.5.5.2 Trajectory Requirements

a. Parking orbit - The LV shall insert the CSM at cutoff into the particular mission design trajectory for the parking orbit within the accuracies defined by the following flight parameters:

Eccentricity, e,	±0.00725
Semi major axis, a,	±10.70 nautical miles
Ascending node,	±0.210 degrees
Inclination,	±0.0839 degrees
True anomaly of the ascending node (If applicable for the particular mission)	±0.830 degrees

b. Missions - The LV shall inject the CSM at cutoff into the particular mission design trajectory within the accuracies defined by the following orbital parameters:

Eccentricity, e,	±0.*
Semi major axis, a,	± * nautical miles
Ascending node,	± * degrees
Inclination,	± * degrees
True anomaly of the ascending node (If applicable for the particular mission)	± * degrees

*MSC shall assign values to these parameters that are consistent with the targeting requirements of a lunar mission and with the V budget established for midcourse correction (Ref. 3.1.3.3)

3.1.5.6 Launch Vehicle Mechanical Interfaces.-

3.1.5.6.1 Physical Interfaces

a. CSM SIVB adapter/IU interface - The CSM SIVB Adapter shall structurally and functionally adapt the SM to the LV. In the area of interface with the LV, design of the Adapter and the design of the IU shall meet the requirements of ICD's 13M20108 (Saturn IB) "Instrument Unit to Spacecraft Physical Requirements" (original issue), 13M50103 (Saturn V) "Instrument Unit to Spacecraft Physical Requirements" (original issue). Requirements established by ICD 13M50123 "Envelope, LEM/SIVB/IU Clearance, Physical" (original issue) will be met as required for the Saturn missions involved.

Note: While the effectivity of these documents is limited to missions A201 and A501, the design requirements established therein provide a baseline reference for all Block I Saturn IB and Saturn V missions.

b. "Q" ball to CSM interface - The design of the "Q" ball and the design of upper end of the ballast enclosure shall meet the requirements ICD's 13M20109 (Saturn IB) "Spacecraft/"Q"-Ball Physical Requirements" (original issue) and 13M50112 (Saturn V) "Spacecraft/"Q"-Ball Physical Requirements" (original issue).

Note: While the effectivity of these documents is limited to missions A201 and A501, the design requirements established therein provide a baseline reference for all Block I Saturn IB and Saturn V missions.

3.1.5.6.2 CSM SIVB Adapter/IU Interface Compartment.-

a. Boost phase venting - During the boost phase the SM, Adapter, IU and SIVB forward skirt shall be vented to atmosphere via vents to be located on the SIVB between 122 and 130 inches aft of the Adapter/IU interface. Total vent cross-sectional area shall be 200 square inches.

b. Purge requirements - Provisions shall be included in the design of the vehicle for a gas purge of the Adapter/IU interface compartment. The purge gas shall be introduced through the umbilicals in the SM, Adapter and IU and shall be exhausted via the boost phase vents in the SIVB. The design shall be compatible with an air purge for the control of temperature and working conditions inside the compartment and with a GN_2 purge when an explosion hazard potentially exists.

c. Lower adapter access provisions - Provisions shall be incorporated in the design of the IU for installation of platforms required for access to the lower LEM area and lower Adapter mounted CSM equipment during ground checkout and servicing operations. These platforms are to be designed and provided by the MSFC. These platforms shall also be designed such that they will provide selected base (leg) attach points and will support the vertical loads only from auxiliary two-man platforms.

3.1.5.7 <u>Launch Vehicle Electrical Interfaces</u>.-

3.1.5.7.1 <u>Adapter/IU Interface Provision</u>.- Three type PTOOSE-24-61S electrical connectors shall be provided in the Adapter for electrical mating of CSM and launch vehicle. The connectors shall be mounted to the adapter approximately 25 inches above the SLA/IU interface, and approximately 45 degrees from the -Z axis toward the + Y axis (CSM axes, Table I).

3.1.5.7.2 <u>"Q" Ball Interface Provisions</u>.- Wiring shall be provided from the MSFC-furnished "Q" ball to the Adapter/IU interface. Wiring shall be terminated with an ME414-0095-0062 or equivalent connector at the "Q" ball interface and one of the above type PTOOSE-24-61S connectors at the adapter/IU interface. The interface between launch vehicle equipment and the CSM portion of the launch vehicle EDS related to the "Q" ball signals is contained in 3.1.5.7.5.2.1, a, (3), (f).

3.1.5.7.3 <u>Power Interface</u>.- Electrical interfaces between CSM and launch vehicle shall be designed in such a manner that there will be no exchange of electrical power between CSM and launch vehicle.

3.1.5.7.4 <u>Signal Interfaces</u>.- For electrical signal interfaces, adequate electrical isolation shall be provided in the interface design so that the effectiveness of any signal crossing the interface will not be deteriorated.

3.1.5.7.5 <u>Launch Vehicle Emergency Detection System (LV-EDS)</u>.- The LV-EDS is a system which is operative during boost flight in both the Saturn LV and the CSM system. Its purpose is to detect critical conditions arising from malfunctions within the LV and automatically transmit a signal to the LES to initiate abort action or to provide information to the CSM crew to indicate that an abort may be required. This specification is concerned only with that portion of the LV-EDS which is contained in the CSM (hereafter referred to as LV-EDS) and its relationship with the portion of the LV-EDS which is contained in the Saturn LV. Where reference is made to the LV portion of the LV-EDS, it is so indicated. Since the CSM system will be engaged in missions involving both Saturn IB and Saturn V launch vehicles, the performance and interface requirements for both those vehicles are included in this specification. These requirements are common to both vehicles except where indicated otherwise.

3.1.5.7.5.1 Performance Requirements.- The LV-EDS contained in the CSM system will provide the following capabilities.

3.1.5.7.5.1.1 Astronaut Displays.- In response to signals from the LV and other CSM subsystems, the LV-EDS will display critical conditions to the crew of the CSM.

3.1.5.7.5.1.2 Astronaut Controls.- The CSM will incorporate provisions for astronaut control of the LV-EDS. These controls will permit the astronaut to:

a. Switch power to or from the LV-EDS system.

b. Manually enable the automatic abort circuitry in the event of failure of automatic enabling at lift-off.

c. Manually initiate an abort sequence with the LES, or, after LES jettison, the SM propulsion subsystem and, concurrently, command LV active engine cutoff.

d. Manually deactivate the entire automatic abort mode by a single switch.

e. Manually deactivate the automatic abort signal for angular overrates for all three planes simulataneously with one switch.

f. Manually deactivate the first stage 2-engines-out automatic abort signal.

3.1.5.7.5.1.3 Automatic LV-EDS Functions.- The LV-EDS will also automatically accomplish the following functions within the CSM.

a. Enable the automatic abort circuitry at the instant of lift-off.

b. Determine through majority voting logic the validity of an automatic abort signal presence in the CSM/LV interface abort circuitry before transmitting an abort command to the LES.

c. Provide an indication to the LV-GSE of an "unsafe" condition in the LV-EDS prior to lift-off. Unsafe being defined as: "attempting to command an abort to the LES which would produce abort action if the automatic abort circuitry were enabled at that time."

d. Disable, in the CSM, the automatic abort circuitry at the instant of LES separation.

e. Initiate the LES abort and shut down the engines on active LV stage upon a valid command from the CSM/LV interface abort circuitry.

3.1.5.7.5.2 Functional Interfaces.-

3.1.5.7.5.2.1 Launch Vehicle Interfaces.- Interchange of LV-EDS signals between the CSM and the LV will be as shown below. The power source to operate these signals shall be as indicated:

a. Launch vehicle to CSM signals

(1) Automatic abort circuitry - Loss of power in two out of three of the CSM/LV interface automatic abort circuits shall cause abort action to be taken by the LES. CSM power shall be used for this circuitry.

(2) Automatic enabling of auto abort circuitry at lift-off - A dual redundant signal from the LV-IU to the CSM shall cause the automatic abort circuitry in the CSM to be enabled, i.e., to be switched into a state of operational readiness. CSM power shall be used for this circuitry.

(3) Display circuits - CM displays will be activated as shown below on receipt of signals from the LV. These display circuits are normally de-energized prior to signal transmittal.

(a) Engine status signals - A discrete signal from the LV-IU to the CM will indicate the nonthrusting status of each of the active LV engines. Eight signal paths will be provided on Saturn IB missions for use during SIB stage burn and one signal path for SIVB stage burn. On Saturn V missions, five signal paths will be provided for use during SIC and S-II stage burn and one signal path for SIVB stage burn. The signals for the different stages on each vehicle shall utilize common circuitry. CSM power shall be used for this circuitry.

(b) Excessive rate signal - A discrete signal from the LV-IU to the CM will indicate the LV rate limit in any of the pitch, roll or yaw planes has been exceeded. CSM power shall be used for this circuitry.

(c) Launch vehicle guidance failure signal - A discrete signal from the LV-IU to the CM will indicate that the LV guidance system has failed and that attitude control is lost, (Rate control will still be operative). CSM power shall be used for this circuitry.

(d) Abort request signal - A discrete signal from the LV-IU to the CM will indicate that either the Range Safety Officer has transmitted a destruct and engine cutoff command to the LV or that Launch Control Center (LCC) is indicating an abort necessity. CSM power shall be used for this circuitry.

(e) Lift-off signal - A descrete signal will be transmitted from the LV-IU to the CM to indicate that lift-off has occurred. CSM power shall be used for this circuitry.

(f) Angle of attack signal - An analog signal will be transmitted from the "Q"-ball/CSM interface to provide a continuous readout of an aerodynamic parameter which is a function of angle of attack. The signal from the "Q" ball interface will be a single signal representing the combined pitch and yaw vectors to give a total angle of attack function readout. The display parameter will be differential pressure across the "Q"-ball on the LES. LV power shall be used for this circuitry.

(g) S-II stage fuel pressure signal (on Saturn V missions only) - An analog signal will be transmitted from the LV-IU to provide a continuous readout of S-II fuel tank pressure. LV power shall be used for this circuitry.

(h) SIVB stage fuel pressure signal (on Saturn V missions only) - An analog signal will be transmitted from the LV-IU to provide a continuous readout of SIVB fuel tank pressure. LV power shall be used for this circuitry.

(i) S-II stage second plane separation signal (on Saturn V missions only) - A discrete signal from the LV-IU will indicate that S-II second plane separation (S-II aft skirt) has occurred. CSM power shall be used for this circuitry.

b. CSM to LV signals

(1) LV engine cutoff circuitry - An abort command transmitted to either the LES or SM propulsion systems (after LES jettison) will cause an engine cutoff signal to be transmitted from the CSM to the LV. This signal will consist of loss of power in at least two out of three energized circuits crossing the CSM/LV interface. LV power shall be used for this circuitry.

(2) Astronaut manual control circuitry - Upon astronaut initiation of the following functions, signals will transmitted to the LV as indicated.

(a) Two engine out auto-abort disable - When the astronaut commands this function, a signal will be transmitted from the CM to the LV-IU. The interface circuitry shall consist of triple redundant wire paths which become energized when the disabling signal is transmitted. LV power shall be used for this circuitry.

(b) LV excessive rates auto-abort disable - When the astronaut commands this function, a signal shall be transmitted from the CM to the LV-IU. The interface circuitry shall consist of triple redundant wire paths which become energized when the disabling signal is transmitted. LV power shall be used for this circuitry.

(3) LV-EDS unsafe signal - Prior to lift-off, the CM will supply a signal to the LV-IU (for subsequent action in the LV-GSE release ladder to prevent lift-off) in the event the LV-EDS circuitry is in an "unsafe" condition. LV power shall be used for this circuitry.

3.1.5.7.5.2 LV-EDS/CSM SCS/CSM G&N Interfaces.- In addition to the displays which are provided for signals from the LV/CSM interface, the following parameters will be displayed for detection of critical conditions arising from LV malfunctions. These displays will be provided by CSM SCS, the signals for which will be in turn provided by the CSM G&N subsystem (operating in the monitor mode) as described in SID 62-1000, G&N Performance and Interface Specification.

a. Attitude error - A continuous readout of vehicle attitude error shall be provided during first stage burn.

b. Total attitude - A continuous readout of vehicle total attitude shall be provided.

c. Angular rates - A continuous readout of vehicle angular rates in each of the pitch, yaw, and roll planes shall be provided.

3.2 Command and Service Module System Design and Performance Criteria.

3.2.1 General Design Analysis Criteria. - Design and operational procedures shall be conducted in accordance with rational design principles to include but not be limited to the following:

3.2.1.1 Limit Conditions. - The design limit load envelope shall be established by superposition of rationally deduced critical loads for all flight modes. Load envelopes of rationally deduced critical loads for all flight modes. Load envelopes shall recognize the cumulative effects of additive type loads. No subsystem shall be designed incapable of functioning at limit load conditions.

3.2.1.1.1 Ultimate Factor. - The ultimate factor shall be 1.5 applied to limit loads. This factor may be reduced to 1.35 for special cases subject to rational analysis and negotiation with MSC, NASA.

3.2.1.2 Performance Margins. - Rational margins shall be apportioned to subsystems and components such that the greatest overall design efficiency is achieved within the LV capabilities and implementation criteria constraints.

3.2.1.2.1 Multiple Failure. - The decision to design for single or multiple failures shall be based on the expected frequency of occurrence as it affects subsystem reliability and safety.

3.2.1.2.2 Design Margins. - All CSM subsystems shall be designed to zero or positive margins of safety.

3.2.1.2.3 Attitude Constraints. - Attitude control is permissable to eliminate system constraints which would impose successive subsystem requirements.

3.2.1.3 Performance Criteria

3.2.1.3.1 CM Pressurization. - The repressurization subsystem shall be designed for two complete cabin repressurizations and a continuous leak rate as high as 0.2 pounds per hour.

3.2.1.3.2 Vacuum Operation of Cabin Equipment. - Vacuum, for design criteria purposes, shall be defined as follows:

For CSM exterior:	7.5×10^{-10} mm hg
For CM interior:	10^{-4} mm hg
For SM interior:	10^{-6} mm hg

3.2.1.3.3 CM Reuse. - The CM and internal subsystems shall not be designed for repeated mission reuse after recovery.

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3.2.1.3.4 CM Flotation and Water Stability. - CM flotation and water stability characteristics shall be such that the CM will recover from any initial attitude and will float upright with normal egrees hatches clear of the water.

3.2.1.3.5 Portable Life Support System (PLSS) Recharge. - Facilities shall be provided for resupply of PLSS expendables. Expendables for resupply of the PLSS expendables shall not be provided.

3.2.1.4 Command and Service Module Design Criteria. -

3.2.1.4.1 Thermal Control. - Thermal design of the S/C modules shall normally use passive means of thermal control, such as insulation, coatings, and control of thermal resistances. Full cognizance shall be taken of thermodynamic considerations in establishing conceptual design and selection of propellants, working fluids, materials for all subsystems such that the maximum allowable temperature range consistent with other design considerations shall be provided. Thermal design of the S/C shall take advantage of change in orientation to the extent possible without compromise of required operational flexibility.

3.2.1.4.2 Maintenance. - Equipment arrangements, accessibility, and interchangeability features that allow efficient preflight servicing and maintenance shall be given full consideration. Design considerations shall also include efficient mission scrub and recycle procedures.

3.2.1.4.3 Ground Handling. - Full design recognition shall be given to the durability requirements of CSM equipment and subsystems during preflight preparation.

3.2.1.5 Environmental Criteria. - These requirements define the environmental design criteria for the CSM equipment and associated Ground Support Equipment (GSE).

3.2.1.5.1 CSM and GSE Ground Environments. - The following conditions represent the natural and induced environmental extremes which may be encountered during transportation, ground handling and storage. Handling GSE shall be capable of operating during exposure to these environments. Other GSE and CSM equipment may be protected by suitable packaging for transportation and storage if these environments exceed the equipment design operation requirements.

3.2.1.5.1.1. Natural environments.

(a) Temperature

Air transportation	-45 to + 140 F for 8 hours
Ground transportation	-20 to + 145 F for 2 weeks
Storage	+25 to + 150 F for 3 years

(b)	Pressure	
	Air transportation	Minimum of 3.47 psia for 8 hours (35,000 ft. altitude).
	Ground transportation and storage	Minimum of 11.78 psia for 3 years (6,000 ft. altitude)
(c)	Humidity	0 to 100 percent relative humidity, including conditions wherein condensation takes place in the form of water or frost for at least 30 days.
(d)	Sunshine	Solar radiation of 360 Btu per square foot per hour for 6 hours per day for 2 weeks.
(e)	Rain	Up to 0.6 inch per hour for 12 hours, 2.5 inch per hour for 1 hour.
(f)	Sand and dust	As encountered in desert and ocean beach areas, equivalent to 140-mesh silica flour with particle velocity up 500 feet per minute and a particle density of 0.25 grams per cubic foot.
(g)	Fungus	As experienced in Florida climate. Materials will not be used which will support or be damaged by fungi.
(h)	Salt spray	Salt atmosphere as encountered in coastal areas, the effect of which is simulated by exposure to a 5-percent salt solution by weight for 48 hours.
(i)	Ozone	Up to 3 years exposure to 0.05 parts per million concentration or 3 months at 0.25 ppm or 72 hours at 0.5 ppm.
(j)	Ground winds	These ground wind criteria consist of a description of Cape

Kennedy wind data for the height intervals of 10 to 400 feet.

(1) Free standing - The design wind speeds for structural loading considerations of the CSM are presented in the table below. Wind speed occuring during the strongest wind month at Cape Kennedy, Florida, are less than those presented 99.9 percent of the time.

Height (ft)	Steady State Wind (knots)	Peak Wind (knots)(*)
10	23.0	32.2
30	28.7	40.2
60	32.9	46.1
100	36.5	51.1
200	41.9	58.7
300	45.4	63.6
400	48.1	67.3

(*) Gust Characteristics:

For the effects of gusts, a linear buildup from the steady state winds to the peak winds will be assumed. The period of this buildup and decay shall be taken as 4 seconds for all height levels; that is, build-up of 2 seconds and 2 seconds for decay to steady state wind speed.

(2) Storm conditions - The 99.9 percent peak wind speeds presented in 3.2.1.5, (1), (j), 1. may be exceeded during severe thunderstorm or hurricane condition at Cape Kennedy. During such periods, the vehicle must be protected in such a manner that wind loading conditions greater than those for the 99.9 percent winds shall not be experienced by the CSM.

3.2.1.5.1.2 Induced environment. -

(a) Shock - as experienced in any direction

Weight (pounds)**	Shock Level (g)	Time (milliseconds)
Less than 250	30	11 ± 1 (half-sine waveform)
250 to 500	24	11 ± 1 (half-sine waveform)

500 to 1,000	21	11 ± 1 (half- sine waveform)
Over 1,000	18	11 ± 1 (half-sine waveform)

**Weight of equipment and package or containers (if any).

(b) Vibration, - Sinoidal as experienced in any direction

Weight (pounds)**	5 to 26.5 cps	26.5 to 52 cps (inch DA)	52 to 500 cps
Less than 50	±1.56 g	0.043	±6.0 g
50 to 300	±1.30 g	0.036	±5.0 g
300 to 1,000	±1.30 g	0.036	
Over 1,000	±1.04 g	0.029	

** NOTE: Weight of equipment and package or containers, if any.

3.2.1.5.1.3 Sheltered environment areas. - These requirements represent the environment design criteria for CSM equipment and GSE both in operating and non operating conditions. The equipment shall be capable of meeting the operating requirements of the applicable performance specification during and after exposure to these environments. Natural and induced environments are combined in this section. The level of environmental control at each Apollo site shall be as indicated in MSC-GSE-1B.

3.2.1.5.1.3.1 Interior controlled. - An environment in which the temperature, humidity, sand, salt spray, etc., are controlled.

(a) Temperature	+60 to +80 F for up to 3 years. +52 to +105 F for 1 hour maximum with environmental equipment out of commission
(b) Oxygen atmosphere	The following conditions apply to the CM interior: 95 ± 5 percent by weight oxygen at total pressures up to 14.7 psia for up to 24 hours. Oxygen partial pressure up to 14.7 psia coincident with total pressure up to 21.0 psia for 2 hours.

(c) Humidity	30 to 70 percent for up to 3 years.
(d) Sand and dust	Particle count not to exceed Level 300,000 of Federal Standard 209: No more than 2,000 particles per cubic foot larger than 5 microns. No more than 35 of these larger than 65 microns. No more than 3 of these 35 particles larger than 100 microns. The following conditions apply to the CM interior, and during open fluid systems activity: Particle count not to exceed Level 100,000 of Federal Standard 209: No more than 700 particles per cubic foot larger than 5 microns. No more than 35 of these larger than 20 microns.

3.2.1.5.1.3.2 <u>Interior uncontrolled</u> - An environment in which the temperature, sand, salt spray, etc., are only partially controlled.

(a) Temperature	+15 to +105 F for up to 3 years
(b) Humidity	0 to 100 percent relative humidity, including conditions wherein condensation takes place in the form of water or frost for at least 30 days.
(c) Sunshine	Solar radiation at 360 Btu per square foot per hour for 6 hours per day for 2 weeks.
(d) Sand and dust	As encountered in desert and ocean beach areas, equivalent to 140-mesh silica flour with particle velocity up 500 feet per minute and a density of 0.25 grams per cubic foot.

(e) Salt spray — Salt atmosphere atmosphere as encountered in coastal areas, the effect of which is simulated by exposure to a 5-percent salt solution by weight for 48 hours.

3.2.1.5.1.3.3 Other environment areas.- Enviornments to which certain GSE are exposed, such as the launch umbilical tower and environmental chamber, shall be as indicated in MSC-GSE-1B.

3.2.1.5.2 CSM Flight Environments. - These requirements represent in environmental design criteria for the CSM equipment in an operating condition as experienced during the various flight mission phases. The mission phases are as defined in 3.1.4.1.1. The equipment shall be capable of meeting the operating requirements of the applicable performance specification during and after exposure to these environments.

3.2.1.5.2.1 General. - These are induced environments which are present for all mission phases.

(1) Temperature — The contractor shall provide temperature requirements for structure, subsystem, and component design for each applicable mission phase.

(2) Oxygen atmosphere — The following conditions apply to the CM interior: 95 ± 5 percent by weight of oxygen for 336 hours. nominal CM interior atmospheric composition is presented in the following table:

Constituent gas	Partial Pressure (psia)	% By Vol.	% By Wt.
Oxygen	4.638	92.76	93.49
Carbon dioxide	0.147 (max)	2.94	4.07
Water vapor	0.215	4.30	2.44

(3) Humidity

The following conditions apply to the CM interior:
0 to 100 percent relative humidity for 336 hours.
40 to 70 percent nominal relative humidity.
95 $\pm$ 5 percent relative humidity including conditions where condensation takes place in the form of water, for at least 20 hours.

(4) Corrosive contaminants

The following condition applies to the CM interior:
Salt atmosphere as caused by human perspiration, the effect of which is simulated by exposure to a 1 per-cent salt solution by weight for 48 hours.

3.2.1.5.2.2 <u>Ascent phase</u>. -

3.2.1.5.2.2.1 <u>Natural environments</u>

(a) Reference Atmosphere

The reference Earth Atmosphere at Cape Kennedy, Florida, shall be in accordance with MSFC Memorandum R-AERO-Y-12-63.

(b) Ground winds

The design wind speed for launch of the CSM are presented in the table below. Wind speeds occuring during the strongest wind month at Cape Kennedy, Florida, are less than those presented 95.0 percent of the time.

Height (ft)	Steady State Wind (knots)	Peak Wind (knots)*
10	14.0	19.6
30	17.5	24.5
60	21.0	29.4
100	22.5	31.5
200	25.9	36.3
300	28.0	39.2
400	29.4	41.2

For the effects of gusts, a linear buildup from the steady state winds to the peak winds will be assumed. The period of this buildup shall be taken as 4 seconds for all height levels; that is, buildup of 2 seconds and 2 seconds for decay to steady state wind speed.

(c) Winds aloft - CSM — Design shall consider a 99 percentile wind shear (Figure 15) with a 9 meter per second discrete quasi-square wave gust super-imposed, such that the total does not exceed a 88 percentile wind speed. See figure 16.

3.2.1.5.2.2.2 Induced environments. -

(a) Pressure — The following condition applies to the CM interior: 14.7 psia nominal decreasing to 6.0 psia. Cavity pressures shall be ambient ± 1 psi.

(b) Vibration — Mechanical vibration from all sources of excitation as experienced by the CSM structure. The design vibration levels for various zones of the CSM are presented in Figures 17 through 21.

(c) Acoustics — Acoustic noise resulting from ground reflection and aerodynamic turbulence. The design acoustics levels for various zones of the CSM are presented in Figures 22 through 34.

(d) Acceleration — The design sustained acceleration levels for the CSM are presented in Figures 35 through 38.

(e) Aerodynamic heating — The design shall utilize the trajectory shown in Figure plus LES plume imfringement where appropriate.

3.2.1.5.2.3 Earth parking orbit, translunar injection, translunar coast, lunar orbit insertion, lunar operations, transearth coast, and pre-entry phases

3.2.1.5.2.3.1 <u>Natural Environments.</u> -

(a) Electromagnetic — The source of electromagnetic radiation presented below impinge on the exterior of the CSM in logical combination for a total time up to 336 hours.

Solar flux	442 Btu/ft^2-hr
Earth emission	73 Btu/ft^2-hr
Lunar emission (subsolar point)	387
Lunar emission (dark side)	5 Btu/ft^2-hr
Earth albedo	0.34

Average lunar maria normal albedo	0.065
Average lunar continents normal albedo	0.105
Average lunar normal albedo	0.073

Space sink temperature zero degrees R

(b) Micrometeoriod — The meteoroid environments for near-earth, cislunar, and near lunar space that shall be used for spacecraft design is as follows:

[redacted]

Near-Earth, cislunar, and near lunar sporadic meteoroids

Flux, mass: $\log N = -1.34 \log m - 10.423$
Where N = number of impacts per square foot per day
m - mass in grams

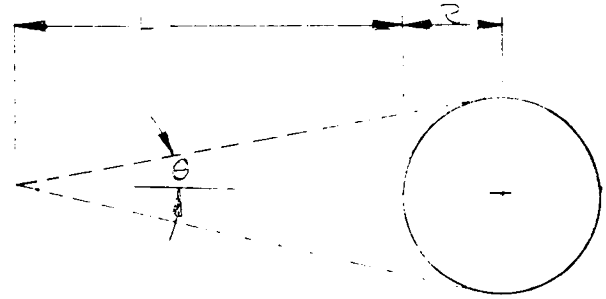
Density: 0.5 gm/cc, all particle sizes

Average geocentric velocity: 30 km/sec, all particle sizes

The flux-mass relationship stated above is shown graphically in Figure 66. All logarithms are to base 10. The flux relation given above is an average of the monthly variations. The above sporadic criteria are applied to the surface area of the vehicle.

Viewing loss $= \frac{1 - \cos \theta}{2}$

Where $\sin \theta = R/R+H$
R = radius of shielding body
H = distance from surface of shielding body



3.2.1.5.2.3.2 Induced environments. -

(a) Pressure

Location	Pressure	Max. Exposure Time
CSM exterior	7.5×10^{-10} mmHg	336 hours
CSM interior (SM and CM forward and aft compartments)	1.0×10^{-6} mmHg	336 hours
CM interior	6.0 psia decreasing to 5.0 psia	(parking orbit only)
	5.0 ± 0.2 psia (normal)	336 hours
	1.0×10^{-4} mmHg (emergency)	100 hours

(b) Vibration — The design levels for the CSM are presented in Figures 39 and 40.

3.2.1.5.2.4 Entry Phase

3.2.1.5.2.4.1 Natural environments. -

(a) Pressure	The following condition applies to the CM interior: 5.0 psia increasing to 5.5 psia (nominal)
(b) Vibration	The design vibration levels for the CM are presented in Figure 19 when uniformly reduced by 10 db.
(c) Acceleration	The design sustained acceleration level is 20 g.
(d) Aerodynamic heating	The design shall utilize the trajectories described in 3.2.2.2.2(f).

3.2.1.5.2.5 Recovery phase

3.2.1.5.2.5.1 Natural Environments

(a) Reference atmosphere	Same as 3.2.1.5.2.d, (1)
(b) Sea state	These conditions are based on 95 degrees cumulative percent frequency winds and wave conditions at the primary landing sites and 90 cumulative percent frequency at the contingency landing sites for the worst month of the year.

Wind velocity	3 to 28.5 knots
Wave height	.5 to 8.5 feet
Wave slope	0 to 8.4 degrees

3.2.1.5.2.5.2 Induced environments. -

(a) Pressure	The following conditions applies to the CM interior: 5.5 psia (nominal) increasing to 14.7 psia (nominal)

(b) Shock — Terminal peak saw-tooth pulse of 78 g (peak amplitude) with total duration 10 to 15 milliseconds, including decay time no greater than 10 percent of the total duration. Figures 41 and 42 define the accelerations.

3.2.1.5.2.6 Launch aborts. -

3.2.1.5.2.6.1 Natural environments

(a) Reference atmosphere — Same as revised 3.2.1.6.2.b(1)(a).

(b) Ground winds for ascent — Same as 3.2.1.6b(1)(b).

(c) Winds aloft — Same as 4.2.1.6.2B(1)(c).

(d) Sea state — These conditions are based on 95 cumulative percent frequency winds and wave conditions at launch abort landing sites for owrst month of the year.

Wind velocity	25.0 knots
Wave height (crest to trough)	11.0 feet
Wave slope	8.7 degrees

(e) Land — These conditions are for launch abort landing sites at Cape Kennedy, Florida

Wind velocity	23.7 degrees

Soils (static bearing strength and slope) 6,000 lb/ft^2 with up to 15-degree slope (loose sand)

25,000 lb/ft^2 with up to 5-degree slope (hard sand)

3.2.1.5.2.6.2 Induced environments. -

(a) Vibration	Mechanical vibrations from all sources of excitation as experienced by primary structures. The design vibration levels for various zones of the CSM are presented in Figures 43 and 44.
(b) Acoustics	Acoustic noise resulting from aerodynamic turbulence and the launch escape motor. The design acoustics levels for various zones of the CSM are presented in Figures 45 and 50.
(c) Acceleration	The design sustained acceleration level is 20 g.

3.2.1.5.3 Command Module Post Landing Environments. - These requirements represent the environmental design criteria for CM equipment in an operating and a nonoperating condition. Operating equipment is that needed for CM habitability and location. This equipment shall be capable of meeting the operating requirements of the applicable performance specification during exposure to these environments for 48 hours.

a. Natural Environment

(1) Temperature

ambient air	85 degrees F maximum up to 48 hours.
sea	85 degrees F maximum up to 48 hours.
(2) Altitude	Sea level
(3) Ambient humidity	85 percent maximum at 85 degrees F. ambient air temperature, up to 48 hours
(4) Solar radiation	306 BTU/ft.^{2} - hr. maximum for a maximum of 6 hrs. with linear increase and decrease to/zero BTU/ft.^{2} - hr. in 5 hrs.
(5) Rain	Same as 3.2.1.6.1,a, (1), (f).

(6) Sand and dust — Same as 3.2.1.6.1,a, (1), (f).

(7) Salt spray — Same as 3.2.1.6.1, a, (1), (h).

(8) Sea state

	First 48 hours	Next 5 days
Wind velocity	3-28.5 knots	3-40 knots
Wave height (crest to trough)	.5-8.5 feet	.5-18 feet
Wave slope	0-8.4 degrees	----------

3.2.1.6 Weights. - The weight of the CSM shall be minimum consistent with design requirements and shall not exceed the control weight of 21,200 pounds for the CSM at launch excluding SPS usable propellant of 40,525 pounds.

3.2.1.6.1 Control Weights. -

Command Module	11,000
Service Module (1)	10,200
Total Command and Service Module (1)	21,200
S-IVB Adapter (2)	3,900
Launch Escape System (3)	8,200

NOTES:
(1) Excluding usable SPS propellant.
(2) Includes 100 pounds structural member for non-LEM operations.
(3) Includes boost protective cover and ballast.

Individual Command and Service Module weights may be varied as required within the limiting Command and Service Module Control Weights shown and as further constrained by any other weight critical component (s) or subsystem (s) subject to NASA concurrence.

CO[redacted]

3.2.1.6.2 Government-Furnished Equipment. - The following GFE items and associated weights are those used in the above control weights.

GFE Total		(1346.1
Guidance and Navigation		430.0
Crew Systems		779.3
Crew (50-70-90)	528.0	
Spacesuits (3)(Incl. Suit Mtd Comm)	90.8	
Survival kit	68.1	
Food set for 10.6-day mission	73.6	
Drinking water probe	0.5	
Medical equipment	4.2	
Bioinstrumentation	3.8	
Radiation dosimeters	5.0	
Constant wear garments	5.3	
Instrumentation (R&D)		34.5
PAM/FM/FM package	16.0	
Gas chromatograph	9.5	
Commutators (3)	9.0	
Scientific equipment		80.0
"Q" Ball (LES)		22.3

3.2.2 Structural Subsystem. - The CSM structural subsystem shall be comprised of the fundamental load-carrying structures. Meteroid and radiation protection shall be that inherent in the structure designed to carry the loads.

[redacted]

3.2.2.1 Subsystem Requirements. -

3.2.2.1.1 Structural Loads. - The primary structure of all modules of the SC shall be capable of withstanding all loads resulting from the conditions specified below without requiring pressure stabilization.

a. Launch phase - Primary S/C structures are to be designed for loads during launch as specified in 3.1.1.3

b. Entry phase - Primary CM structures are to be designed for a limit load of 20 g during entry.

c. Noise - The design shall accommodate sound pressure levels in the respective frequency ranges shown in Figures 22 through 34 and 45 through 50.

d. Vibration - The effects of the steady and transient inputs shall be combined. The vibration analyses shall recognize the lower damping present in a vacuum. The vibration curves are shown in Figures 17 through 25 and 37, 40, 43, and 44.

e. Dynamic loading - The calculation of dynamic loads shall include the effects of engine start, rebound on the pad, lift off transients including ground winds, gusts, wind shears, boost, engine shutdown, and separation.

f. Abort - The LEV shall be designed to oscillating and tumbling abort conditions except that LES jet impingement pressures will be limited to the structural capability compatible with other loading conditions.

g. Land and Water Landing - Structural deformation is allowable within the limits of crew safety. Water leakage into the crew compartment of the CM after impact shall be such that there is no impairment of post loading functions (uprighting recovery communications, post landing ventilation, etc.) during a 48-hour period with no crew corrective action. The contractor shall take corrective action to eliminate failure acodes resulting in leakage that is identified in the development and qualification program.

3.2.2.1.2 Pressure Vessel. - Pressure vessels shall be designed with the following considerations:

a. Pressure vessel limit loads - Limit loads shall be obtained with limit pressures. When pressure effects are relieving, pressure should not be used. Limit pressure is defined as the relief valve nominal pressure plus its tolerances and plus hydrostatic head.

b. Pressure vessel ultimate factor - The ultimate factor shall be 1.50.

c. Pressure vessel proof factor - The proof factor shall be 1.33 when pressure is applied as a singular load.

3.2.2.2 Subsystem Description. -

3.2.2.2.1 Launch Escape Tower. - The tower structure shall form the connecting link between the CM and the structural skirt of the launch escape motor, and shall be designed to carry the loads and stresses to which it will be subjected in performing its function of aborting the CM at any point from the launch pad to 30 seconds after ignition of the Saturn S-II. The four main longitudinal members shall terminate at the CM, forming a rectangular pattern. Attachment of each of these four members to the CM shall be by means of explosive bolts, which shall function to detach the tower structure from the CM at the initiation of the jettison command. The launch escape tower shall be protected by an ablative material to prevent overheating.

3.2.2.2.2 Command Module. - The CM physical features shall be defined by aerodynamic and heating performance requirements and crew utility and well being considerations.

a. Geometric characteristics - The basic external geometry of the CM is shown in Figure 51. The CM shall be a symmetrical, blunt body developing a minimum hypersonic L/D of 0.30. The L/D vector shall be effectively modulated in hypersonic flight by the use of roll control.

b. Inboard profile - Basic arrangements of internal features fundamental to full utilization of the CM geometry shall be as shown in Figures 52, 53, and 54.

(1) Load attenuation swept volume - The crew shall be suspended on discrete load attenuation devices which normally act on Earth-landing impact.

(2) Crew cabin interior - All equipment and structures within the crew cabin shall be free of sharp protrusions which constitute a hazard to the crew or crew equipment.

c. Center of gravity management - Consideration shall be given to center of gravity management. Alteration of crew positions may be used for center of gravity management after touchdown.

d. Visibility - Visibility shall be provided by one window over the head of the crew in the launch condition. Four additional windows compatible with temperature requirements of the lunar mission shall be provided for use during the flight phase.

e. Access and egress hatches -

(1) There shall be one side hatch provided in the CM to be used for ground access servicing and maintenance. Normal access and egress for the crew and all on-board equipment installation shall be achieved through the side hatch. The capability shall exist for unaided egress of the crew on the pad with the boost protective cover installed. To be accomplished within 90 seconds.

(2) There shall be another inward opening hatch at the forward end of the crew compartment for possible egress after landing.

f. Entry thermal protection - The CM shall be designed with a thermal protection shell which will insure that the internal environment of the CM will not exceed the design limits of the structure and its enclosed system while entering the Earth's atmosphere. The entry thermal protection system shall be designed for entry within a nautical mile corridor with a maximum deceleration of 10 g during the intitial pull out with a nominal L/D - 0.50 with angle of attack tolerance of $\pm$ 4 degrees at a entry velocity of 36,200 fps. The maximum range at nominal L/D shall be 5,000 nm. The maximum emergency decleration limit shall not exceed 20 g.

g. Inner structure - The pressure cabin shall be separate from the thermal protection subsystem. The space between the pressure cabin and thermal protection shell shall be vented to limit the collapsing pressures. No provision shall be made for over-pressures due to LV explosion except that due to existing structural capability.

3.2.2.2.3 Service Module. - The SM shall be designed and constructed to support body loads from the SM and Adapter and provide a mounting structure for: SM subsystems, pressure vessels for EPS and ECS reactants, and the attachment for the high gain S-Band Antenna. Space radiators shall be in integral part of the SM outer shell. The SM reference axes are delineated in Table I.

a. Inboard profile - The SM internal arrangement shall contain six segments and a center section. (see Figures 55 and 56.) Equipment contained in each sector shall be as follows:

(1)	Sector I	Cryogenic tankage and smaller items of other equipment
(2)	Sector II	OPS oxidizer tank
(3)	Sector III	SPS fuel tank
(4)	Sector IV	Fuel cells, SPS pressurization package, and smaller items of other equipments
(5)	Sector V	SPS oxidizer tank
(6)	Sector VI	SPS fuel tank
(7)	Center section	SPS helium storage tanks (two)

The high gain S-band antenna shall be housed below the lower SM bulkhead and inside the adapter. The antenna shall be extended after adapter - SM separation.

b. SPS tank sizing - Tank sizing for the SPS shall provide a minimum usable propellant storage for 30,000 pounds of oxidizer and 15,000 pounds of fuel.

3.2.2.2.4 CSM/SIVB Adapter. - The CSM SIVB adapter shall structurally and functionally adapter the SM to the launch vehicle. In the area of interface with the launch vehicle, design of the adapter shall meet the requirement of ICD's 13M20108 (Saturn IB) "Instrument Unit to Spacecraft Physical Requirements" (original issue), 13M50103 (Saturn V) "Instrument Unit to Spacecraft Physical Requirements" (original issue). Requirements established by ICD 13M50123 "Envelope, LEM/SIVB/IU Clearance, Physical" (original issue) will be met as required for the Saturn missions involved.

a. Adapter separation - The adapter shall be designed so that separation of the adapter from the CSM shall be effective by severing the adapter shell with ordnance devices at a station below the SM/S-IVB adapter interface. Simultaneously, longitudinal explosive charges shall further separate the shell into hinge panel segments. The lower section of the Adapter shell will be left intact and attached to the S-IVB booster.

3.2.3 Guidance and Navigation Subsystem (G&N). - This subsystem shall be provided by NASA and shall be consistent with the Guidance and Navigation subsystem P&I Requirements Specification, SID 62-1000.

3.2.4 Stabilization and Control Subsystem (SCS). -

3.2.4.1 Subsystem Requirements. - The SCS shall provide for:

a. Angular orientation and stabilization of the spacecraft by controlling CM and SM RCS engine firing.

b. Manual translational control along the three major S/C axes by controlling SM RCS engine firing.

c. Thrust vector control during powered phases of the mission by commanding the gimbal angle and thrust of the SM SPS.

d. Direct manual control of the CM and SM reaction control subsystem.

e. Backup attitude reference.

f. Display of body rates, body attitude errors, and total CM attitude.

g. The subsystem shall accept commands from the guidance subsystem or crew for thrust vector control, entry control, and attitude control.

h. Accept discreet control commands and attitude change commands from the Apollo programmer for unmanned flights.

3.2.4.2 Subsystem Description.- The SCS shall consist of the following basic components:

Attitude reference
Rate sensors
Control electronic assembly
Manual control
Displays
Power supplies

3.2.5 Service Propulsion Subsystem (SPS). -

3.2.5.1 Subsystem Requirements. - The SPS shall supply the propulsion increments in the following normal and emergency modes.

a. All major velocity increments required for translunar midcourse velocity corrections, for insertion of the CSM into a lunar orbit, for lunar orbit into the transearth trajectory, and for transearth midcourse velocity corrections.

b. Abort propulsion after jettison of the LES.

3.2.5.1.1 Propellants and Pressurant. - The SPS shall utilize the following fluids:

a. Nitrogen tetroxide (N_2O_4) as the oxidizer.

b. A mixture of 50 percent hydrazine (N_2H_4) and 50 percent unsymmetrical dimethylhydrazine (UDMH) as the fuel.

c. Helium (He) as the pressurant.

3.2.5.1.2 Performance. - The subsystem shall have the following performance characteristics.

a. Thrust = 21,500 pounds nominal in a vacuum.

b. Mimimum specific impulse: Isp = 313 seconds (-36 valve) at end of 750 seconds of operation."

c. Operating life = 750 seconds minimum.

d. Minimum impulse bit = 50000 $\pm$ 200 pounds second.

3.2.5.2 Subsystem Description. - The SPS shall consist of the following components:

3.2.5.2.1 Rocket Engine Subsystem. - The SPS engine shall be a single unit, liquid-fueled, pressure-fed, non-throttleable trust generator, gimbal-mounted to permit thrust vector control with a maximum gimbal angle of $\pm$8.5 degrees in the X-Y plane and $\pm$6.0 degrees in the X-Z plane with multiple restart capability.

3.2.5.2.2 Propellant Subsystem. - The propellant subsystem shall consist of an oxidizer and a fuel supply, each with a storage and sump tank in series, and a distribution system.

3.2.5.2.3 Pressurization Subsystem. - The pressurization subsystem shall consist of a high-pressure helium supply, contained within two spherical tanks, and associated pressure regulators, isolation valves, check valves, and pressure relief valves.

3.2.5.2.4 Propellant Quantity Gaging. - A propellant quantity gaging assembly shall be provided. This gaging assembly in conjunction with displays shall provide quantity remaining data to the crew.

3.2.5.2.5 Propellant Utilization. - A manually operated propellant utilization control shall be supplied.

3.2.6 Reaction Control Subsystem (RCS). - The CSM shall include reaction control subsystems to provide the impulse for attitude control and stabilization. The SM/RCS shall also be capable of minor translational velocity increments.

3.2.6.1 Command Module Reaction Control Subsystem (CM/RCS). - This subsystem shall be used only after separation of the CM from the SM.

3.2.6.1.1 Subsystem Requirements. - The subsystem shall provide three-axis control prior to development of aerodynamic moments, roll control during entry, and pitch and yaw damping during entry. A minimum impulse bit of not more than 2 pound seconds shall be provided. The subsystem shall have the capability to delete unused propellant and pressurant prior to impact. The subsystem shall also be capable of providing three axes during the control fo the CM all high altitude aborts.

3.2.6.1.2 Subsystem Description. - The CM/RCS shall be pulse-modulated, pressure-fed, and utilize earth storable hypergolic propellant. Two separate subsystems shall be provided which are capable of independent or simultaneous operation, and each shall be capable of meeting the total torque and impulse requirements. Each subsystem shall consist of pressurization and propellant storage/distribution sections and six thrusters installed to provide three-axis rotational control. Propellant tanks shall be positive expulsion type. Each subsystem shall have the capability for manual isolation of the pressurant and propellant sections.

a. Oxidizer - The oxidizer shall be nitrogen tetroxide (N_2O_4)

b. Fuel - The fuel shall be monomethylhydrazine (MMH).

c. Pressurant - The pressurant shall be helium (He).

3.2.6.2 Service Module Reaction Control Subsystem (SM/RCS). - This Subsystem shall be used only after CSM separation from the launch vehicle.

3.2.6.2.1 Subsystem control. - The SM RCS shall provide translational and rotational control of the CSM during all unpowered phases and roll control during powered phases. The subsystem shall be capable of supplying a minimum impulse bit of 0.4 $\pm$ 0.2 pound-seconds. The subsystem shall also be capable of providing the impulse required for the following maneuvers:

a. CSM separation from the launch vehicle.

b. Ullage settling of SPS propellant.

c. Minor velocity corrections of less than SPS capability."

3.2.6.2.2 Subsystem Description. - The RCS shall be pulse-modulated, pressure-fed, and utilize earth-storable hypergolic propellants. The subsystem shall consist of pressurization and propellant storage/distribution sections and 16 thrusters installed to provide 3-axis rotational and translational control. Propellant tanks shall be positive expulsion type. A propellant quantity indicating section shall be provided which is functional in a zero gravity environment. The subsystem shall have a capability for manual isolation of the pressurant and propellant sections.

a. Oxidizer - The oxidizer shall be nitrogen tetroxide (N_2O_4).

b. Fuel - The fuel shall be a mixture of 50 percent hydrazine (H_2H_4) and 50 percent unsymmetrical dimethylhydrazine (UDMH).

c. Pressurant - The pressurant shall be helium (He)."

3.2.7 Launch Escape Subsystem (LES). - A LES shall be provided. -

3.2.7.1 Subsystem Requirements. - The LES shall be capable of separating the CM from the LV in the event of failure or imminent failure of the LV on the launch pad and during all atmospheric phases.

3.2.7.1.1 Performance Criteria. - The LES shall provide for crew escape from a critically malfunctioning LV from the time of access arm retraction until shortly after the second booster stage separation. Crew accelerations incurred during LES abort and entry following abort shall not exceed the emergency limits of Figures 57 through 59.

a. Jettison Capability - Propulsion and trajectory shaping for LES jettison shall be provided.

b. Crew Escape - The LES shall provide for crew escape from the LV under the following conditions:

(1) For escape prior to and shortly after lift off. The LES shall separate the CM from the LV and propel the CM to an adequate altitude to ensure safe recovery operation. The minimum no-wind range requirement for aborts from a nominal launch vehicle (zero attitude change and zero degree per second attitude rate) shall be 3,000 ft. at apogee. The plane of the abort trajectory shall be nominally in a down range direction.

(2) The abort capability shall provide for critical launch vehicle malfunctions which occur at lift-off. A nominally performing subsystem shall provide recovery at, or above, ground levle for the following booster malfunction conditions and associated parameters for abort initiation.

	Average Booster Divergence Rate Deg/Sec	Attitude Divergence at Abort Initiation Deg.
Condition I		
Condition II		

(3)

(4) During approximately the first 40 seconds following lift-off, range safety considerations preclude thrust termination.

(5) Approximately 40 seconds after lift-off, the LV thrust will be terminated automatically at abort initiation.

(6) A minimum separation of the CM from the LV at maximum dynamic pressure shall be 350 feet in 3 seconds following CM separation from the SM. For all aborts, a minimum "miss distance" of 800 feet shall be provided for the case of abort at zero degrees angle of attack and stable flight of the LV. The term "miss distance" is defined as the dis-

(7) The LES shall be capable of performing its function at the maximum dynamic pressure incurred during the boost (see Figure 3), with abort initiated prior to structural break-up of the LV or spacecraft configuration. A minimum capability shall be provided for abort at conditions described as follows:

Altitude Dynamic Pressure	= 40,000 feet = 750 psf
Condition I (Slow divergence failure)	Attitude rate at abort initiation abort = 5 degrees per secind, angle of attack or side-slip at abort initiation of ±15 degrees
Condition II (Hard over-gimbals)	Average pitch (or Yaw) acceleration prior to abort initiation of 10 degrees per second^2. Pitch (or yaw) rate at abort initiation of ±5 degrees/second.

(8) The maximum altitude for LES abort shall exceed the altitude for:

(a) Completion of second stage ignition and separation of jettisoned components

(b) Achieving a dynamic pressure condition permitting utilization of a SM abort.

3.2.7.1.2 <u>Normal Mission LES Jettison</u>. - A jettison capability shall be provided to separate the LES from the boosters. A sufficient lateral separation distance shall be provided to assure a minimum "Miss-distance" of 150 feet when jettison is initiated from a nominal space vehicle. For combined worst off-nominal conditions of the space vehicle and the LES, the LES shall be capable of achieving separation andavoiding recontact with the space vehicle.

3.2.7.2 <u>Subsystem Description</u>

3.2.7.2.1 <u>Launch Escape Subsystem (LES)</u>. - The LES shall include the following components:

a. Q-Ball

b. Launch escape tower canard system

c. Pitch control motor

d. Launch escape motor

e. Structural skirt

f. Tower structure

g. Tower/CM separation system

h. Tower jettison motor

i. Forward heat shield separation and retention system (Not included in control weight for LES.)

j. Boost protective cover

k. Tension ties (Not included in control weight for LES)

3.2.7.2.2 Abort Initiation and Control. - Abort shall be initiated manually or automatically by the LV-EDS. Following abort initiation, LES functions shall be automatically controlled by the automated sequence control subsystem. Manual control of physical functions shall be provided to enhance reliability of the subsystem.

CO[redacted]

3.2.8 Earth Recovery Subsystem (ERS). - The CM shall include an ERS to be used under all flight conditions for earth landing requirements.

3.2.8.1 Subsystem Requirements. - The subsystem shall satisfy the following requirements after normal entry, and pad escape or launch abort.

3.2.8.1.1 The parachute system shall be designed for loads resulting from CM gross launch weight of 11,500 pounds with a factor of safety of 1.35.

3.2.8.1.2 Recovery Stabilization. - Stabilize the CM during recovery.

3.2.8.1.3 Velocity Control. - Reduce the vertical touchdown velocity to not more than 35.0 feet per second at sea level.

3.2.8.1.4 Impact Attenuation. - Reduce impact acceleration such that the CM flotation is not impaired. Any further attenuation required to prevent exceeding the crew emergency acceleration limits delineated in Figures 57 through 59 shall be provided by crewman shock attenuation devices. The CM pitch attitude at impact is nominally negative 26 degrees.

3.2.8.1.5 Postlanding. - The ERS will provide as auxiliary equipment on the CM the following equipment:

a. Provisions for recovery antenna deployment.

b. Provisions for visual location aids.

c. Provision for CM flotation attitude control.

d. Exterior recovery party communications umbilical connection.

e. CM pick up sling.

3.2.8.2 Subsystem Description.- The ERS shall consist of two actively reefed ribbon type drogue parachutes deployed by mortar and a cluster of three simultaneously deployed, actively reefed Ringsail landing parachutes, crushable honeycomb ribs in the CM impact area, and crushable honeycomb shock struts for crew couch impact attenuation.

Drogue parachutes shall be sized so that any one shall stabilize the vehicle descent, and in the proper attitude for landing parachute deployment when required.

Main parachutes shall be sized so that satisfactory operation of any two of the three will satisfy the vertical velocity requirement. Pyrotechnic devices to disconnect the drogue and landing parachutes shall be provided.

[redacted]

3.2.9 Crew Subsystem

3.2.9.1 Subsystem Requirements. - Design and operational procedures shall be in accordance with the crew requirements presented here.

3.2.9.1.1 Crew Size and Number. - The CSM design parameters shall accommodate three crew members between the 10th and 90th percentile, as defined in WADC-TR 52-321, Anthropometry of Flying Personnel, for the following dimensions: weight, standing height, sitting height - erect, buttock-to-knee length, knee height (sitting), hip breadth (sitting), shoulder breadth (bideltoid), and arm reach from wall. All other body dimensions shall fall within the 5th and 95th percentiles as defined by WADC-TR 52-321. Percentiles for body dimensions undefined by applicable documents will be estimated.

3.2.9.1.2 Division of Duties. - Division of duties shall be as follows:

a. Crew duty requirements shall be based on cross-training so that each crew member is able to perform tasks performed by other crew members.

b. Tasks will be apportioned to make efficient utilization of all crew members.

c. There will be an established order of command within the crew.

d. S/C design will recognize the principle distinction in crew duties and designations.

3.2.9.1.3 Metabolic Requirements. - The average daily metabolic requirements for each crew man are assumed to be as shown in Table II.

3.2.9.1.4 Environmental Requirements. - The CM interior environment shall be as specified in 3.2.10.

3.2.9.1.5 Decompression Protection. - Pressurized garments (GFE) shall provide protection for crew members in the event of crew compartment decompression. Two crew members shall be capable of donning pressure garment assemblies in 5 minutes or less without assistance. At least one crew member shall wear the pressure garment assembly at all times.

3.2.9.1.6 Food and Water

a. Food - Provisions for storage of food and associated equipment (GFE) shall be provided within the CM sufficient for a 14-day mission.

b. Water - In addition to the primary source of potable water, a backup supply shall be provided in the survival kit (GFE). Chemicals capable of desalting sea water shall be made available. One pint of fresh water shall be obtained from each 0.16 pounds of chemical.

3.2.9.1.7 Human Waste Control. - Provisions shall be provided for the removal and disposition of gaseous, solid (fecal), and liquid human waste within the CM. A manually operated valve shall be provided for periodically venting liquid waste overboard.

3.2.9.1.8 Portable Light. - A portable light shall be provided for illumination of the CM interior.

3.2.9.2 Subsystem Description

3.2.9.2.1 Crew Equipment. - Provisions for the crew equipment are delineated in SID 62-1003, Preliminary NASA Furnished Crew Equipment interface and performance specification and the Exhibits for the Apollo Block I space suit assembly procurement package.

3.2.9.2.2 Couches. - Couches shall be designed to provide comfortable support during all mission phases. All three crew couch seat pans shall fold to the extent required, to provide necessary work space and adequate access by the crew to all regions of the CM as required.

3.2.9.2.3 Restraint Subsystem. - A subsystem of restraints shall be provided for crew support and restraint during normal and emergency mission conditions. The restraints shall be designed to the acceleration limits in Figure 57, 58 and 59 and impact limits.

3.2.9.2.4 Crew Accessories. - All crew accessories shall be provided to assist the crewmen in the performance of tasks under anticipated mission conditions and activities. SID 62-1003, Preliminary NASA-Furnished Crew Equipment Interface Requirements specifies the NASA furnished crew equipment.

3.2.9.2.5 Window Filter Assemblies. - An opaque window shade assembly shall be provided for all S/C windows to attenuate solar heat and visible radiation.

3.2.9.2.6 Survival Kit Storage. - Provision shall be made for storage and accessibility of three sets of survival provisions (GFE) in the CM.

3.2.9.2.7 Crew Equipment and Suit Interface. - A subsystem of umbilicals and connectors shall provide electrical power and oxygen circulation for the pressure garment assembly. This equipment shall also include tethering provisions for tools used under weightless conditions. Provisions shall be made for stowage of two PGA helmets and glove pairs allowing rapid access for emergency donning.

3.2.9.2.8 Medical Kit Storage. - Provision shall be made for storage of one GFE medical kit in the CM.

3.2.9.2.9 Personal Hygiene. - Personal hygiene equipment shall be provided to enable crewmen to perform necessary body cleansing during the mission.

3.2.9.3 NASA-Furnished Crew Equipment

3.2.9.3.1 Survival Provisions. - Survival equipment is to be provided which will support and aid in the location and rescue of the crewmen during the post landing situation of 48 hours maximum duration. Climatic conditions do not include extreme cold. Medical requirements shall be satisfied by retrieval of the S/C medical kit. Contents of the survival kit shall be GFE and shall be packaged in the contractor-furnished survival package in the CM.

The NASA-furnished survival provisions shall be packaged in the contractor-furnished survival provisions assembly for storage in the CM.

3.2.9.3.2 Personal Equipment. - The NASA-furnished personal equipment shall include:

a. Pressure garment assembly
b. Constant wear garment (CWG)
c. GFE checkout equipment, as required

3.2.9.3.3 Medical Equipment. - The NASA-furnished medical equipment shall include the following items:

a. Radiation dosimeter (Set)
b. Dressings - emergency medical kit (Set)
c. Medications - emergency medical kit (Set)
d. Instrument set - clinical monitoring, physiological
e. Instrument assembly - biomedical preamplifier
f. Instrument assembly - biomedical sensors, personal

Items a through d of the NASA-furnished medical equipment shall be stored in the contractor-furnished medical compartments aboard the CM.

3.2.9.3.4 Food and Associated Equipment. - The NASA-furnished food and associated equipment shall consist of the following items:

a. Food
b. Mouthpiece, food, personal
c. Probe, water delivery

The NASA-furnished food shall be stored in the contractor-furnished food compartment assembly for storage in the CM.

3.2.10 <u>Environmental Control Subsystem (ECS)</u>. - The CSM shall include an ECS which provides a conditioned, "shirtsleeve" atmosphere for the crew; provisions for space suits in event of cabin decompression; thermal control of all CSM equipment where needed; and provisions only for charging the Portable Life Support System (PLSS).

3.2.10.1 <u>Subsystem Requirements</u>

3.2.10.1.1 <u>Cabin Pressure</u>. - The cabin pressure nominal operating limits shall be 5 $\pm$ 0.2 psia. The subsystem shall be capable of maintaining a cabin pressure of at least 3.5 psia for at least 5 minutes following a single 1/2-inch diameter puncture in the pressure compartment.

3.2.10.1.2 <u>Oxygen Partial Pressure.</u> - The oxygen partial pressure nominal limits shall be 233 millimeters Hg and emergency conditions 160 mm Hg minimum.

3.2.10.1.3 <u>Carbon Dioxide Partial Pressure</u>. - The CO_2 partial pressure nominal limit shall be 7.6 mm Hg maximum. In an emergency the limits shall not exceed that given in Figure 60. In the post-landing phase, a maximum of 16 mm Hg CO_2 concentration shall be allowable.

3.2.10.1.4 <u>Metabolic Action Requirements</u>. - The ECS shall be designed in accordance with the requirements of Table II with the exception of the post-landing phase. For post-landing the following criteria shall be used:

a. Metabolic rate - 800 Btu/hr. per man maximum.

b. Sweat rate - As specified in Figure 61.

c. Drinking water - 12.4 lb/day-man.

d. CO_2 production - 3.6 lb CO_2/man-day.

e. Allowable effection temperatures - specified in Figure 62.

3.2.10.1.5 <u>Temperature Limits</u>

a. CM temperature - The cabin air temperature nonstressed limits shall be 75 $\pm$ 5 degrees F minimum and 80 degrees F maximum; The nominal and emergency limits shall be as presented in Figure 63 and 64 respectively. For the post-landing phase the effective temperature shall be in accordance with Figure 65.

b. SM temperature - The SM temperatures shall be maintained within safe limits for equipment installed.

3.2.10.1.6 Cabin Relative Humidity. - The cabin relative humidity non-stressed limits shall be 40 percent minimum and 70 percent maximum. The nominal and emergency limits shall be as presented in Figures 63 and 64 respectively.

3.2.10.2 Subsystem Description. - Environmental control shall be accomplished with a pressure suit circuit, collant circuit, pressure and temperature control section, oxygen supply section, water management section and a waste management section.

3.2.10.2.1 Pressure Suit Circuit. - This loop supplies the conditioned atmosphere to the cabin and space suit and shall provide removal of debris and noxious gases and for carbon dioxide absorption. Ventilation flow at 3.5 psia and 50 degrees F.10 cfm through each space suit with a maximum flow resistance in each space suit of 5 inches of water.

3.2.10.2.2 CM Pressure & Temperature Control Section. - This section shall provide cabin ventilation, pressurization and thermal control during all phases of the mission.

a. Temperature

Exposure time *(hr)	Max. air temp (F)
0 - 2.5	84.4
2.5 - 8	84.4 linear increase to 86.5
8 - 12	86.5
12 - 17	86.6 linear decrease to 84.4
Repeat	

b. Humidity — 82 percent relative humidity for 48 hours

c. Sunshine

Exposure time *(hr)	Solar radiation (Btu/ft^2/hr)
0 - 5	0 linear increase to 306
5 - 9	306
9 - 13	306 linear decrease to 0
13 - 24	0

*Starting at sunrise.

d. Sea State

Wind velocity	3 to 28.5 knots
Wave height (creast to trough)	0.5 to 8.5 feet

3.2.10.2.3 Oxygen Supply Section. - The primary gas supplies shall be stored as super critical cryogenics in the SM in the same tank as for the EPS. Entry oxygen shall be supplied.

3.2.10.2.4 Coolant Circuit. -

a. CM thermal control - Dissipation of the thermal load of the CSM shall be accomplished as required by absorbing heat with a circulating coolant and rejecting heat from a space radiator and water boilers. Other cooling subsystems shall supplement or relieve the primary subsystem.

b. SM thermal control - Thermal stabilization of the SM/RCS shall be accomplished by absorbing and dissipating heat with a circulating coolant by the use of a heat sink, electric heater and pump assembly independent of CM thermal control.

3.2.10.2.5 Water Management. - Water shall be collected from the pressure suit circuit and the fuel cell and stored in positive expulsion tanks. The water collected from the fuel cell shall be sotred separately and used as the primary source of potable water. Water shall be provided at liftoff to satisfy the crew post-landing metabolic needs and provide for evaporative cooling during exit and re-entry following an immediate abort. A water management program shall be encompassed in the design to provide water requirements for all other phases of the mission.

3.2.10.2.6 Waste Management. - Waste management shall provide ventilation for the refuse storage compartments, and the control of gaseous, solid and liquid wastes from within the CM.

3.2.10.2.7 Preflight Checkout. - Fittings with proper access shall be provided to perform pressure checks, components performance tests, etc., of breaking system integrity for component tests. Provision shall be made for testing and calibrating all environmental sensors.

3.2.11 Electrical Power Subsystem (EPS).

3.2.11.1 Subsystem Requirements. - The EPS shall be designed to store energy, generate, supply, regulate, condition, and distribute all electrical power required by the CSM for the full duration of the mission, including the post-landing recovery, but excluding the PLSS batteries.

3.2.11.1.1 Power Output. - The EPS shall be capable of generating 575 kwh of electrical energy from fuel cells at a minimum rate of 563 watts and a maximum rate of 1,420 watts per fuel cell module. In addition, 3,480 watt hours from storage batteries shall be available.

3.2.11.1.2 DC Bus Voltage. - Electrical power shall be generated and distributed at 28 vdc (nominal).

3.2.11.1.3 AC System Voltage. - The ac system shall supply 115/200 volts at 400 cps and shall be three phase Y connected.

3.2.11.1.4 Load Grouping. - All electrical loads supplied by the distribution system shall be classified as essential, nonessential, pyrotechnic, or recovery. Essential loads are defined as those loads (except pyrotechnic circuits) that are mandatory for safe return of the CSM to earth from any point in the lunar mission. Loads not necessary for the safe return of the CSM shall be grouped on a non-essential bus and provision made for disconnecting these loads as a group under emergency conditions. All loads required during the post-landing recovery period shall be supplied by a recovery bus and provision made for manually disconnecting the essential bus following landing. Redundant buses shall be provided for pyrotechnic circuits and used to supply only that type load.

3.2.11.1.5 Cryogenic Gas Storage. - Hydrogen and oxygen reactants for the fuel cells shall be stored in the supercritical state in insulated tanks maintained at constant working pressure.

3.2.11.2 Subsystem Description

3.2.11.2.1 Major Components. - The EPS shall include the following major components:

a. Energy sources

   Cryogenic gas storage section
   storage batteries

b. Power generation equipment

   fuel cell section

c. Power conversion equipment

   Inverters
   Battery chargers

d. Power distribution equipment

   Power buses, a-c and d-c
   Associated controls

3.2.11.2.2 Location. - The location of each of the above components within the CSM shall be as listed herein. Every effort shall be exercised to minimize equipment size and weight, commensurate with the established requirements and obtaining the highest practicable reliability.

		Location
Fuel cell modules, radiators, and controls		SM
Cryogenic tanks (empty), piping, valves	SM	SM
Total reactants, plus reserves		SM
Entry batteries		CM
Pyrotechnic batteries		CM
Separation sequencer batteries		SM
Battery charger		CM
Static inverters		CM
EPS display and control panel		CM

3.2.11.2.3 Operating Modes

3.2.11.2.3.1 Normal Operation. - During all mission phases, from launch until reentry, the primary electrical power requirements of the CM-SM shall be supplied by three fuel cell modules operating in parallel. The storage batteries described in 3.2.11.1.1 may be utilized to supply required power above the normal capacity of the three fuel cell modules for short duration peaks. Batteries shall be fully charged prior to reentry.

In the event a failure occurs to one of the fuel cell modules, the failed unit shall be capable of being electrically and mechanically isolated from the subsystem and the electrical load required to continue the mission shall be assumed by the two fuel cell modules remaining in operation. The storage batteries may be utilized to supply required power above the normal capacity of two fuel cell modules for short duration peaks. Batteries shall be fully charged prior to reentry.

3.2.11.2.3.2 Emergency Operation. - In the event of a failure to two fuel cell modules, the failed units shall be capable of being electrically and mechanically isolated from the subsystem. S/C electrical loads shall be immediately reduced by the crew and manually programmed to remain within the generating capabilities of the one remaining operable fuel cell module. The storage batteries may be utilized to supply required power above the normal capacity of one fuel cell module for short duration peaks. Batteries shall be fully charged prior to reentry.

3.2.11.2.3.3 Entry and Recovery. - The fuel cell modules and SM accessories will be jettisoned with the SM. All subsequent electrical power requirements shall be provided by the CM storage batteries.

3.2.12 Communications Subsystem

3.2.12.1 Subsystem Requirements. - The communications subsystem shall provide the capabilities listed below and shall be compatible with the CSM/MSFN interfaces as specified in SID 64- CSM/MSFN Communications Interface Specification.

a. Provide for two-way voice communications between crew members inside the CSM.

b. Provide for S-band transmission of voice, PCM telemetry, and television signals to the MSFN from the CSM at lunar distances.

c. Provide for S-band transmission of recorded PCM and analog data to the MSFN from the CSM at lunar ranges.

d. Provide for coherent S-band pseudo random voice ranging between the CSM and MSFN at lunar distances.

e. Provide for emergency voice and key S-band transmission to the MSFN from the CSM at lunar ranges.

f. Provide for VHF/FM PCM transmission to the MSFN from the CSM during near-earth phases.

g. Provide a television camera capable of generating high resolution, 10 frame per second, television pictures for S-band transmission to the MSFN.

h. Provide for reception of S-band voice and up data from the MSFN at lunar distances.

i. Provide for reception of UHF up data from the MSFN during near-earth phase.

j. Provide for transmission and reception of VHF/AM voice communications between the MSFN and CSM during near-earth phases.

k. Provide a time reference for the CSM communications subsystem and for use by other CSM subsystems. This time reference is to synchronized with the CSM G&N subsystem. Provide for MSFN up dating of this time reference.

l. Provide data processing for conditioning and encoding CSM analog and digital data into a PCM format for VHF/FM and S-band transmission

m. Provide for recording and playback of PCM and analog data.

n. Provide for two-way VHF/AM and HF voice communications with recovery forces.

o. Provide for VHF and HF direction finding by recovery forces.

p. Provide for pulsed C-band radar tracking by the MSFN during near-earth phase.

q. Provide for two-way HF voice communications with the MSFN during extended orbital missions when beyond line-of-sight ofVHF and S-band transmission.

3.2.12.2 System Description. - The major components of the Block I communications subsystem shall include the following items:

a. VHF/FM transmitter equipment

b. VHF/AM transmitter-receiver equipment

c. Unified S-band equipment

d. S-band power amplifier equipment

e. C-band transponder equipment

f. VHF recovery beacon equipment

g. HF transceiver equipment

h. Audio center equipment

i. Television equipment

j. PCM telemetry equipment

k. Premodulation processor equipment

l. Digital up-data link equipment

m. Signal conditioning equipment

n. Central timing equipment

o. Data storage equipment

p. Beacon antenna equipment

q. VHF/2-KMC omni antenna equipment

r. Recovery antenna equipment

s. Anxillary equipment

t. 2-KMC high gain antenna equipment

u. Flight qualification recorder

v. HF orbital antenna equipment

3.2.12.3 GFE. - The following GFE equipment shall be provided:

a. VHF survival beacon

b. Flight qualification recorder calibrator - timer.

3.2.12.4 Functions. - The communication subsystem functions and detailed description are contained in SID 54-1237, Vehicle Model Specification, Basic Block I.

3.2.13 <u>Instrumentation Subsystem</u>.

3.2.13.1 <u>Subsystem Requirements.</u> - The CSM shall contain an instrumentation subsystem capable of sensing the operational status, environmental, and performance parameters of various subsystems. The instrumentation subsystem shall be capable of converting and doncitioning the sensed data and providing a signal for display, telemetry, recording or checkout purposes. The parameters to be sensed are those required by the crew for normal operation, abort operation and decision making for the control of the CSM. Also, parameters will be sensed and processed that will provide data to ground personnel to verify the basis for and participate in major decision-making processes of the flight crew.

3.2.13.2 <u>Subsystem Description</u>. - The operational instrumentation will be physically and functionally compatible with the other subsystems noted in SID 64-1237, Vehicle Model Specification-Basic, Block I. The R&D instrumentation shall be as noted in SID 62-1001, Flight R&D Instrumentation Performance and Interface Specification. Space provisions only for scientific instrumentation will be provided. The space allocation shall consist of 2.7 cubic feet as delineated in SID 64-1237, Vehicle Model Specification-Basic, Block I.

3.2.14 <u>Displays and Controls (D&C)</u>.

3.2.14.1 <u>Subsystem Requirements</u>. - Sufficient depth of information and command access to the CSM subsystems shall be provided to enable the three-man crew to accomplish the following operations:

(a) Effect manual CSM system management and/or control

(b) Safe shutdown of CSM equipment

(c) Select alternate subsystem operating modes

(d) Recognize hazard to crew CSM, launch vehicle or mission and effect mission change if normal system operation cannot be restored.

3.2.14.2 <u>Subsystem Description</u>. - The Display and Control (D&C) subsystem shall present information to and accommodate control action inputs from the CSM flight crew during the mission as described in 3.1.4.

The primary location of the D&C equipment shall be the main display console, which is located above the crew couches in the CM. Secondary locations of the equipment shall include the right-hand and left-hand side display consoles. Other locations of the D&C equipment shall include the left-hand forward equipment bay, right-hand forward equipment bay and the navigation station at the lower equioment bay.

The operational D&C equipment shall be orientated as to establish a D&C subsystem. The master caution and warning subsystem and crew compartment lighting equipment are part of the D&C equipment and shall be provided as defined in SID 64-1237, Vehicle Model Specification, Basic - Block I.

3.2.15 Entry Monitor System.- No requirement exists for Block I mission. Mission definition and operational usage will be employed to provide a method for safe entry.

3.2.16 Sequential Events Control Subsystem.

3.2.16.1 Subsystem Requirements.- Sequencing shall be employed to control those functions and events which require greater precision or speed of response than the crew can provide or to relieve the crew of tedious tasks.

3.2.16.2 Subsystem Description.- The sequencing subsystem description is delineated in detail in SID 64-1237, Vehicle Model Specification-Basic, Block I. This subsystem shall be capable of performing the proper sequencing of events during ascent, entry, LES Abort, Adapter separation and SPS abort, initiating functions and providing monitor capabilities.

3.2.17 Pyrotechnic Subsystem and Devices.

3.2.17.1 Subsystem Requirements.- All Pyrotechnic Subsystem electrical circuitry shall provide for redundant design through the electro-explosive interface.

3.2.17.1.1 Standard Electro-Explosive Device (EED).- All electrically-actuated pyrotechnic devices shall be fired by the Apollo standard initiator (ASI).

3.2.17.1.2 Standard Detonator Cartridge.- A standard detonator shall be used to initiate all high explosive charges. The detonator shall consist of the ASI hermetrically sealed into a cartridge containing a charge which produces a high, order detonation. This cartridge assembly shall be designated as the Apollo standard detonator (ASD).

3.2.17.1.3 Electrical Power Sources.- The firing and logic electrical power sources for pyrotechnic subsystems shall be provided by batteries which are separate and independent from all other CSM power sources and from each other.

3.2.17.1.4 Firing Circuit Tests.- Provisions shall be made for electrical continuity checkout of all firing circuits after mating of the last electrical connector in the circuit. Special test equipment shall be used for this continuity test to preclude dangerous levels of voltage being inadvertently applied.

3.2.17.1.5 Secondary Effects.- Possible adverse effects of the operation of explosive devices, such as fragmentation of sharp edges, shall be minimized.

3.2.17.1.6 Circuit Isolation.- The firing circuits including power sources for pyrotechnic subsystems shall be separate and independent from all other CM circuitry.

3.2.18 Service Module Propellant Dispersal Subsystem. - This pyrotechnic function shall be operable upon receipt of the RF arm and fire signals initiated by the ground range safety officer. The subsystem shall be operable only during that portion of flight prior to LES tower jettison. The explosive charges shall open the SM main propellant tanks to provide atmospheric dispersal of propellants. Space provisions only shall be provided for explosive charges and devices to disperse contents of the LEM descent stage propellant tanks.

3.3 Reliability Requirements

3.3.1 Mission Success Reliability. - The mission success reliability objective for Apollo shall be for a lunar orbital rendezvous (LOR) mission followed by the return to earth of the CSM without exceeding the emergency crew limits, given in the design criteria.

3.3.2 Crew Safety Reliability. - The crew safety reliability objective for the Apollo LOR mission shall be interprete as the probability that the crew shall not have been subjected to conditions greater than the emergency limits given in the design criteria.

3.3.3 Reliability Apportionment. - The reliability objectives for the major Apollo-Saturn systems shall be as delineated below:

Apollo-Saturn Reliability Apportionments

System	Mission Success	Crew Safety
GSE	0.9999	0.99999
MSFN	0.999	0.99999
LAUNCH VEHICLES (defined by the NASA)	0.950	0.99994
CM and SM		
APOLLO-SATURN	0.90	0.999

Reliability apportionment shall be based on a 10.6-day mission.

3.4 Electromagnetic and Electrostatic Compatibility. - Each assembly shall electromagnetically compatible with other assemblies in the system, other equipment in or near the LV, associated test and checkout equipment, and to the electromagnetic radiation of the operational environment. The subsystem shall not be a source of interference that could adversely affect the operation of other equipments or compromise its own operational capabilities. The system shall not be adversely affected by fields or voltages reaching it from external sources, such as improperly suppressed vehicle test and checkout equipment, and nearby radio frequency sources in the operational environment and electrostatic potential.

3.4.1 CSM and GSE Equipments. - MC 999-00-0002B, Electromagnetic Interference control for the Apollo Space System dated January 3, 1963, shall be used as a guide.

3.4.2 CSM and GSE Subsystems. - Shall be designed in accordance with specification MIL-E-6051C, Electrical-Electronic System Compatibility and Interference Control Requirements for Aeronautical Weapon System, Associated Subsystem, and Aircraft, dated 17 June 1960.

3.5 Interchangeability. - Mechanical and electrical interchangeability shall exist between like assemblies, subassemblies, and replaceable parts of operating subsystems (electronic, electrical, etc) regardless of the manufacturer or supplier. Non-operating subsystems such as structure need not comply with this requirement. Interchangeability for the purpose of this paragraph does not mean identity, but requires that a substitution of such like assemblies, subassemblies, and replaceable parts be easily effected without physical or electrical modifications to any part of the equipment or assemblies, including cabling, connectors, wiring, and mounting, and without resorting to selection; however, adjustment of variable resistors and trimmer capacitors may be made. In the design of the equipment, provisions shall be made for design tolerances sufficient to accommodate various sizes and characteristics of any one type of article, such as tubes, resistors, and other components having the limiting dimensions and characteristics set forth in the specification for the particular component involved without departure from the specified performance. Where matched pairs are required, they shall be interchangeable and identified as a matched pair or set.

3.5.1 Identification and Traceability. - Apollo identification and trace-shall be in accordance with the contractor's approved quality control plans.

3.6 Ground Support Equipment (GSE). - GSE is defined as the non-flight implements or devices required to checkout, handle, service, or otherwise perform a function in support of the CSM or boilerplate during tests at factory subsequent to manufacturing complete, prelaunch, and post launch operations at the test site, and major development tests such as house CSM tests, propulsion tests and environmental tests. The Master Ground Operations Specification, SID 63-489, describes specific detailed GSE requirements.

3.6.1 GSE Concept.

3.6.1.1 Design Concept. - The GSE design concept delineates four general categories of equipment for supporting servicing, handling, system checkout and testing, and various auxiliary requirements. The equipment design shall be pointed towards remote control utilizing a digital interface with computer analysis and control as well as a direct interface for local/manual control. To as great an extent as practical, similar equipment shall be used to ensure continuity in checkout. Design shall be based on use by skilled technicians.

3.6.1.2 Operations Support. - CSM GSE shall support the vehicles during: (1) acceptance, (2) test preparation, (3) test, (4) checkout, and (5) prelaunch checkout. It shall also include such recovery and post-launch test items as may be agreed to by the parties.

3.6.1.3 CSM Checkout Concept. - CSM checkout shall consist of:

a. Local/manual operations

b. Remote/semi-automatic operation

c. Remote/manual operation

The local/manual and remote/semi-automatic operations shall be performed with the acceptance checkout equipment-S/C (ACE-S/C - NASA-supplied). Adequate flexibility shall be incorporated to accommodate frequent changes. The equipment required for hazardous operations shall be designed for remote/manual operation from protected areas. NAA shall provide all carry-on equipment for ACE-S/C usage, all flight hardware required for checkout of the CSM with ACE-S/C, and all carry-on for checkout of the Guidance and Navigation subsystem with ACE-S/C.

3.6.1.4 System Checkout Concept. - All checkout operations performed on systems and subsystems installed in the vehicle shall be performed by checkout equipment having remote manual or automatic capability for malfunction detection and isolation to a replaceable package. Operations performed on subsystems not installed in the vehicle may be accomplished by bench test equipment (BTE). BTE shall be limited to local manual operation.

3.6.1.5 Maintenance Concept. - Field maintenance of CSM subsystems shall be performed as follows:

a. For airframe electrical/electronic equipment (either installed or on the bench), checkout and replacement shall be at the integral package (black box) level. A "black box" is defined as a combination of replaceable units which are contained within a physical package.

b. For non-electrical/electronic equipment (either installed or on the bench), checkout and replacement shall be at the lowest replaceable serialized unit level.

BTE may be used for malfunction verification of packages (units) removed from the CSM because of suspected failure. The malfunctioned package (unit) shall be returned to the supplier. BTE may also be used for spares certification before installation.

3.6.2. Support Requirements. - The level to which CSM GSE shall support the vehicles, operations, and sites is as follows.

3.6.2.1. Test Preparation and Acceptance Area. - Equipment shall be provided in the test preparation and acceptance area to functionally checkout S/C subsystems and verify compliance of operational and performance parameters with design requirements. Installation checkout, subsystem functional tests, and integrated systems tests shall be performed. Substitute units shall be provided when required to simulate modules or elements of the system which are not present. Extensive checkout of fuel cell and cryogenic subsystems and associated servicing equipment will not be conducted in this area.

3.6.2.2 House S/C. - GSE for house S/C shall perform tests for the following purposes:

a. Engineering development

b. Field operations checkout (ACE-S/C programming and operations)

The remote manual and semi-automatic checkout modes shall be applied to the house CSM operations to develop checkout techniques and operating procedures as ACE-S/C capabilities are developed. Servicing, handling and auxiliary equipment shall be provided as required.

3.6.2.3. Prequalification Flight Drop Test Site. - The GSE provided to support these operations shall consist of handling equipment, and limited auxiliary equipment.

3.6.2.4. White Sands Missile Range. - Support for the WSMR abort tests shall be in a local/manual mode and remote/manual mode with checkout equipment having capability for centralized gross system and diagnostic check with checkout equipment having capability as specified in 3.6.1.5. The support at WSMR shall include control and monitoring during pre-launch operations and countdown. The R&D instrumentation checkout equipment and associated checkout equipment shall be installed in a mobile van in place of the pad transfer room. The R&D instrumentation checkout equipment shall be modified as necessary to ensure suitability for its intended end purpose. Auxiliary, handling, and servicing equipment shall be provided as required with checkout equipment specified in 3.6.1.3 and 3.6.1.4.

3.6.2.5. Las Cruces Propulsion System Development Facility. - Support of test preparation and firing preparation shall be in the local/manual mode with equipment having diagnostic capability as specified in 3.6.1.5. Servicing equipment required to furnish fluids, propellants, pneumatic pressures for the propulsion, reaction control, and other fluid subsystems shall also be provided with local/manual control capabilities. Engine firing control equipment capable of controlling and monitoring firings in a remote manual mode shall be provided. Handling an auxiliary equipment shall be provided as required.

3.6.2.6. Space Environmental Simulation Laboratory. - Support checkout of the CSM shall be accomplished with ACE-S/C equipment prior to the thermal vacuum test. Handling, and auxiliary equipment shall be provided as required. Test operations will be conducted with remote/manual equipment.

3.6.2.7. Atlantic Missile Range. - Equipment shall be provided for the complete functional checkout of the S/C and verification of readiness for flight. Equipment at the launch complex shall provide for servicing and preparation of the space vehicle and monitoring and control of the launch operation. Special facilities and equipment shall be provided for static firing of the SPS and RCS systems, and operation and verification of the fuel cell, cryogenic and the environmental control systems. Individual subsystem and integrated systems tests shall be conducted in the operations and control building. Tests and checkout in the operations and control building and at the launch complex shall be designed for use of ACE-S/C equipment.

3.6.2.8 Arnold Engineering Development Center (AEDC). - Service Propulsion System tests will be conducted using GSE and STE furnished by the engine subcontractors as well as standard bench type test equipment.

3.6.2.8 Marshall Space Flight Center (MSFC). - To support dynamic and umbilical tests, handling and auxiliary GSE will be utilized at MSFC.

3.7 Personnel Training. - A program plan shall be provided for training the flight crew, ground operations personnel, and other personnel in the skills and knowledge required for operation of the Apollo system. The contractor shall support the program with the following categories of trainers as defined in the subparagraphs hereunder and in SID 64-1807, Apollo Model Specification for Apollo Training Equipment - Block I.

3.7.1 Subsystems Trainers. -

3.7.1.1 Trainer Concept. - The Apollo subsystems trainers shall consist of mechanized lighted-line representations of major CSM subsystems, the controls and displays associated with the subsystems, and trainer controls for the introduction of special conditions and malfunctions. The systems trainers shall be mobile unit devices operating in a controlled environment. The design configuration will be based on the first manned earth-orbital S/C. Subsystems trainers shall provide detailed descriptions of the functions and inter-relationships of components within each subsystem and shall illustrate normal and alternate modes of operation including the effects of malfunctions.

3.7.1.2 Trainer Items. - The Apollo subsystems trainers shall consist of the following major items:

a. Stabiliaation and control subsystem trainer
b. Electrical power subsystem trainer

c. Sequential flow subsystem trainer
d. Environmental control subsystem trainer
e. Propulsion subsystems trainer

3.7.1.2.1 <u>SCS Subsystem Trainer</u>. - The SCS trainer shall provide a functional flow diagram depicting system operation including switching functions. All modes of the SCS shall be demonstratable with automatic, manual, and manual override inputs. An animated pictorial representation shall identify sufficient thrusters on both CSM to illustrate attitude control in the plus and minus direction for the attitude thrust systems. In general, redundant system representation need not be shown.

3.7.1.2.2 <u>Electrical Power Subsystem Trainer</u>. - The EPS trainer shall provide a functional flow diagram depicting dc and ac power generation and power distribution to the main busses. Output from busses to S/C systems need not be detailed. A flow diagram of one fuel cell system including the cryogenic storage system shall be shown. The trainer shall be capable of demonstrating fuel cell operation, inverter operation, battery recharge, bus switching, and system management through use of the S/C panel monitors. Redundant systems and/or controls will not be animated but will appear as static representations. Panel monitors shall show currents, voltages, and other system variables.

3.7.1.2.3 <u>Sequential Flow Subsystem (SFS) Trainer</u>. - The SFS trainer shall portray the sequential operations of the LES, ELS, EDS, and other operations required for crew safety. The trainer shall be capable of demonstrating the sequence of normal launch, pad abort, high altitude abort, and normal earth landing. The representations should begin from appropriate battery busses. System "A" shall be represented completely and System "B" only where required.

3.7.1.2.4 <u>Environmental Control Subsystem (ECS) Trainer</u>. - The ECS trainer shall depict the pressure suit supply system, oxygen supply system with O_2 cryogenic input system (the O_2 cryogenic storage is shown on the EPS trainer), water-glycol system, and the water system. The trainer shall be capable of demonstrating normal operation during all mission phases. Panel monitors shall show pressure and temperature changes for the operating modes.

3.7.1.2.5 <u>Propulsion Subsystem Trainer</u>. - The propulsion trainer shall provide sufficient animated diagrams to depict operation of the CM RCS, the SM RCS, and the SPS. Panel monitors shall illustrate system variable state.

3.7.2 <u>Apollo Part Task Trainer (APTT)</u>

3.7.2.1 <u>Trainer Concept</u>. - The APTT shall provide for the simulation of the S/C with sufficient realism for the training of the flight crew and ground operations personnel in specific mission phase tasks. The trainer shall be a fixed base device operating in a controlled environment and capable of providing training in selected tasks associated with mission segments as follows: launch countdown, launch, earth orbit, translunar, lunar orbit, transearth and entry. The design configuration of the APTT shall be based on the controls and displays of the first manned earth-orbital S/C and related APTT training requirements. Capabilities will be provided in the performance characteristics of the trainer equipment to simulate mission segments objectives of S/C subsequent to the first manned earth-orbital flight including the lunar mission S/C. The APTT shall provide crew training in normal and alternate flight procedures. Malfunctions shall be inserted in the training tasks to require the flight crew to utilize these alternate procedures. Initially, at least one malfunction shall be available for each alternate procedure. Additional malfunctions shall be provided on the basis of subsequent analysis to provide a library of malfunctions related to crew actions and alternatives. Malfunctions for which no crew alternative exists shall not be employed.

3.7.2.2 Major Equipment Groups. - The trainer shall consist of the following major equipment groups:

a. Simulated CM

b. Instructor's console for three instructor positions

c. Computer complex

    Digital computer and peripheral equipment
    Analog computers and peripheral equipment
    Input - output control and buffer unit equipment
    Simulated S/C subsystems equipment
    Aural simulation equipment
    Recording equipment
    Air conditioning equipment

3.7.2.2.1 Simulated CM. - The simulated CM shall be stationary with the X axis vertical. The CM interior as well as the S/C displays and controls shall be simulated to the degree of realism required by the training tasks. Exterior simulation of the CM shall not be required. Reach and visual patterns shall be identical to those in the S/C for the crew couch positions and the navigational sighting station to the degree required by the training tasks. Windows in the CM shall be opaque.

3.7.2.2.1.1 Spacecraft Hardware. - SC Controls, displays and panel designs may be used for taining equipment provided these items are modified as required for trainer requirements. Consideration shall be given to the use of less expensive materials than those required for the S/C and to the less stringent environmental requirements for trainers.

3.7.2.2.2 Instructors' Console. - An instructors' console shall be provided for three instructor operators. The center section of this console shall be similar in layout to the S/C console. Supplementary displays and controls shall be provided on either side of the center section to indicate crew control actions. A malfunction insertion unit shall be provided for the instructor's use.

3.7.2.2.3 Computer Complex. - The computer complex shall provide all computation for the realistic simulation of the mission phases. This complex shall consist of three major equipment subgroups, the digital computer with peripheral equipment, the analog computer with peripheral equipment, and the input-output control and buffer units.

3.7.2.2.4 CSM Subsystem Simulation Equipment. - CSM subsystem simulation equipment shall be provided external to the computer complex to augment the computer equipment in the simulation of the S/C subsystems.

3.7.2.2.5 Aural Simulation Equipment. - The aural simulation equipment shall provide background noise and specific event aural cues to the crew.

3.7.2.2.6 Recording Equipment. - Magnetic tape recording equipment shall be provided for communication and trainer data playback after training sessions.

3.7.2.2.7 Air Conditioning Equipment. - Air conditioning equipment shall provide ambient temperature control in the simulated CM and the pressure suits. Plenumcooling air will be supplied to simulator components by the using facility.

3.7.3 Apollo Mission Simulator. -

3.7.3.1 Trainer Concept. - The Apollo mission simulator (AMS) shall provide for the simulation of the S/C with sufficient realism for the training of the flight crew, ground operations personnel, and integrated flight and ground crews in all phases of the total mission. The trainer shall be a fixed based device operating in a controlled environment and capable of providing training in all tasks associated with continuous mission phases as follows: launch countdown, launch, earth orbit, earth orbit rendezvous and docking, translunar including transposition and docking, lunar orbit, lunar rendezvous and docking, transearth, entry and landing. The phases of the mission shall be presented in a continuous fashion without apparent re-programming or switching transients. The trainer shall be capable of providing integrated training and operation with the manned spaceflight control center (MSCC) and the simulation checkout and training system. The crew and ground operations functions to be trained shall be those encountered in the operational systems being simulated. The design configuration of the AMS shall be based on the controls and displays of the first manned earth-orbital S/C and related AMS training requirements. Growth capabilities shall be provided in the performance characteristics of the trainer equipment to simulate missions of subsequent S/C. The AMS shall provide crew training in normal flight procedures and alternate flight procedures. Malfunctions will be inserted in the training tasks to require the flight crew to utilize these alternate procedures. Initially, at least one malfunction shall be provided for each alternate procedure. Additional malfunctions will be provided on the basis of subsequent analysis to provide a library of malfunctions related to crew actions and alternatives. Malfunctions for which no crew alternative exists will not be employed.

3.7.3.2 Major Equipment Groups. - The trainer shall consist of the following major equipment groups:

a. Simulated CM
b. Instructor-operator station complex
c. Computer complex
d. Simulated SC subsystems equipment
e. Visual simulation equipment
f. Aural simulation equipment
g. Interface equipment
h. Closed circuit television
i. Recording equipment
j. Air conditioning equipment

3.7.3.2.1 Simulated CM. - The simulated CM shall be an authentic replica of the CM internally, with respect to size, shape, and equipment location. The simulated CM shall be stationary with the X axis vertical. Controls and displays shall be authenic replicas of the S/C controls and displays.

3.7.3.2.1.1 Spacecraft Hardware. - S/C controls, displays and panel designs may be used for training equipment provided these items are modified as required for trainer requirements. Consideration shall be given to the use of less expensive materials than those required for the S/C and to the less stringent environmental requirements for trainers.

3.7.3.2.2 Instructor-Operator Station Complex. - The instructor-operator console, which console, which consists of three stations, shall have an arrangement similar to the crew stations of the S/C. The station corresponding to S/C commander shall be master instructor control station. Each station shall have repeater instruments similar to those at the corresponding station in the S/C and other instruments and indicators as required to fulfill the training requirements. Communication facilities shall be provided to permit communications between instructor stations and the crew stations in the CM

3.7.3.2.3 Computer Complex. - Analog computers, digital computers, analog to digital computers and digital to analog converters shall be supplied as required.

3.7.3.2.4 Simulated S/C Subsystems. - The simulated systems shall be realistic representations of actual S/C systems. Characteristics of components of the systems, such as motors, valves, regulators, shall be included, as required, to produce the static and dynamic performance of the systems under simulated normal and malfunction conditions.

3.7.3.2.5 Visual Simulation Equipment. - Simulated external visual stimuli shall be presented to the three crew members through the CM windows, except the hatch window, and the telescope and sextant. Objects viewed through the windows and optical instruments which are pertinent to crew training shall be simulated to a degree of accuracy consistent with the training requirements. The images presented to the windows and optical instruments shall be of such resolution, brightness, distortion level, and accommodation as to be realistic and subjectively acceptable.

3.7.3.2.6 Aural Simulation Equipment. - Aural simulation equipment shall provide background noise and specific event aural cues through loud speakers in the CM and through crew member's headsets.

3.7.3.2.7 Interface Equipment. - Interface equipment shall be provided as required to permit the trainer to operate independently, integrated with the MSCC, integrated with the LEM Mission Simulator (LMS), or integrated with both the LMS and the MSCC.

3.7.3.2.8 Closed Circuit Television. - Television cameras shall be installed in the simulated CM to permit visual monitoring of crews, and the interior of the CM. Camera location shall be out of the crew's normal field of view so far as practicalbe. Sensitivity of the cameras shall be sufficient to present a clear picture to the instructor-operator station complex monitoring scopes, using normal CM interior illumination.

3.7.3.2.9 Recording Equipment. - Magnetic tape recording equipment shall be provided for communication and trainer data playback after training sessions. Analog parameter recording equipment shall be provided for recording of selected dyanmic time histories. Provisions shall be made to permit the selection and recording of those trainer variables indicative of significant characteristics of the training exercise.

3.7.3.2.10 Air Conditioning Equipment. - Air conditioning equipment shall provide ambient temperature control in the simulated CM and the pressure suits, and pressure suit pressurization. Plenum cooling air will be supplied to simulator components by the using facility.

3.8 Manned Space Flight Control Center (MSCC) and Manned Space Flight Net (MSFN). - The design configuration of the MSCC and MSFN shall conform with the CSM-MSFN data flow requirements as specified in SID 64- , CSM/MSFN Communications Interface Specification.

3.9 Materials, Parts, and Processes. - Materials, parts, and processes shall be selected with the following considerations:

a. Materials, parts, and processes shall be suitable for the purpose intended. Safety, performance, reliability, and maintainability of the item are of primary importance.

b. Except in those instances where their use is essential, critical materials shall not be used.

c. Where possible, materials and parts shall be of kind and quality widely available in supply channels.

d. When practicalbe, a choice among equally suitable materials and parts shall be provided.

f. Whenever possible, single source items shall be avoided.

g. When practicable, circuits shall be designed with a minimum of adjustable components.

3.9.1 Specifications and Standards. - Materials, parts, and processes shall be selected in the following order of preference, provided coverage is suitable:

a. Federal specifications approved for use by the NASA

b. Military specifications and standards (MIL, JAN, or MS)

c. Other Governmental specifications

d. Specifications released by nationally recognized associations, committees, and technical societies.

3.9.2 Choice of Standard Materials, Parts, and Processes. - Where applicable, preferred parts lists shall be used. When an applicable specification provides more than one grade, characteristics, or tolerance of a part or material, the standard parts, materials, and processes of the lowest grades, broadest characteristics, and greatest tolerances shall be chosen. However, standard parts, materials, or processes of high grades, narrow characteristics, or small tolerances may be used when necessary to avoid delay in development or production, obvious waste of materials, or unnecessary use of production facilities. The requirements specified for the use of standard parts, materials, or processes shall not relieve the contractor of the responsibility to comply with all performance and other requirements specified in the contract.

MSFC-PROC-158A, 12 April 1962, Soldering electrical connectors (high reliability) - Procedure for, as amended by MSC-ASPO 5B, 18 May 1964, delineating soldering requirements will be complied with.

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3.9.3 Nonstandard Parts, Materials, and Processes. - Nonstandard parts, materials, and processes may be used when necessary to facilitate the design of the particular equipment. However, when such nonstandard items are incorporated in the design, they shall be documented as required by the contract.

3.9.3.1 New Parts, Materials, and Processes. - New parts, materials, or processes developed under the contract may be used, provided they are suitable for the purpose intended. Any new parts, materials, or processes used shall be documented as required by the contract.

3.9.4 Miniaturization. - Miniaturization shall be accomplished to the greatest extent practicable, commensurate with required functions and performance of the system. Minaturization shall be achieved by use of the smallest possible parts and by compact arrangement of the parts in assemblies. Miniaturization shall not be achieved by means that would sacrifice the reliability or performance of the equipment.

3.9.5 Flammable Materials. - Materials that may support combustion or are capable of producing flammable gases (which in addition to other additives to the environment, may reach a flammable concentration) will not be used in areas where the environment or conditions are such that combustion would take place.

3.9.6 Toxic Materials. - Unless specific written approval is obtained from the NASA, materials that produce toxic effects or noxious substances when exposed to CM interior conditions shall not be used.

3.9.7 Unstable Materials. - Materials that emit or deposit corrosive substances, induce corrosion, or produce electrical leakage paths within an assembly shall be avoided or protective measures incorporated.

3.9.8 Fungus-Inert Materials. - Fungus-inert materials shall be used to the greatest extent practicable. Fungus-nutrient materials may be used if properly treated to prevent fungus growth for a period of time, dependent upon their use within the CSM. Whe used, fungus-nutrient materials shall be hermetically sealed or treated for fungus and shall not adversely affect equipment performance or service life.

3.9.9 Metals. - All metals shall be of corrosive-resistant type or shall be suitably protected to resist corrosion during normal service life. Gold, silver, platinum, nickel, chromium, rhodium, palladium, titanium, cobalt, corrosion-resistant steel, tin, lead-tin alloys, tin alloys, Alclad aluminum, or sufficiently thick platings of these metals may be used without additional protection or treatment.

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3.9.9.1 Dissimilar Metals. - Unless suitably protected or coated to prevent electrolytic corrosion, dissimilar metals, as defined in Standard MS 33586, shall not be used in intimate contact.

3.9.9.2 Electrical Conductivity. - Materials used in electronics or electrical connections shall have such characteristics that, during specified environmental conditions, there shall be no adverse effect upon the conductivity of the connections.

3.9.10 Lubricants. - The CSM lubricants and lubrication shall be compatibel with the combined environments in which they are employed. Lubricant material and process specifications will be formulated to prescribe materials and describe application methods.

3.9.11 Special Tools. - The functional components of the CSM and component attachments shall be designed so that the use of special tools for assembly, assembly, installation, and service shall be kept to a minimum.

3.9.12 Hazard Proofing - Propellants and Gases. - The design of the S/C subsystems and support equipment shall minimize the hazard of fire, explosion and toxicity to the crew, launch area personnel and facilities. The hazards to be avoided include accumulation of leakage of combustible gases, the hazard of spark on ignition sources including static electricity discharge, and toxicity due to inhalation or spillage of certain expendables.

Design of equipment shall be in accordance with MSFC 10M01071, during any part of the mission operation. Where practicable, the various components shall be hermetically sealed or of explosion-proof construction. The rocket motor squibs shall be capable of withstanding an electrical impulse of 1 ampere at 1 watt dc for 5 minutes without detonating. Design of equipment shall be in accordance with MSFC 10M01071.

3.9.13 Fail Safe. - Subsystem or component failure shall not propagate sequentially; that is, design shall "fail safe."

3.9.14 Connectors. - Wherever practical, all electrical and mechanical connectors (except R&D instrumentation) shall be so designed as to preclude the possibility of incorrect connection.

3.9.15 Ground Support Equipment. - Commercial standards for materials and equipment shall be utilized to the maximum extent possible where such use will not compromise the safety of operations or the meeting of the necessary performance requirements.

3.9.16 Nameplates and Product Markings. - The CSM and all assemblies, components, and parts shall be marked for identification in accordance with Standard MIL-STD-130.

4. QUALITY ASSURANCE PROVISIONS

4.1 General Quality Assurance Program. - NAA/S&ID shall establish a quality assurance program in accordance with NASA Publication NPC 200-2 and NPC 200-2 and NPC 200-3. Inspections and tests to determine conformance of the system to contract and specification requirements shall be conducted prior to submission of the article to the NASA for acceptance. Documentation requirements shall be a snoted in Exhibit I to the Apollo Contract, NAS9-150. NAA/S&ID shall prepare and submit to NASA a quality assurance program plan per the requirements of Exhibit I.

4.2 Reliability Program. - NAA/S&ID shall establish a Reliability Program in accordance with NASA publication NPC 250-1. Implementation of this document shall be as specified in the NAA/S&ID Reliability Program Plan. (SID 62-203).

4.3 Test. - NAA/S&ID shall establish a qualification test program to determine that the CSM system satisfies the requirements of Section 3 of this specification. The definitions and ground rules for establishing this program are as follows:

4.3.1 Definitions. -

a. Qualification tests - Functional tests performed on production hardware at and above mission levels of all critical environments to assure that the hardware will meet the design requirements and will perform its function for its use cycle.

b. Criticality - Criticality describes the impact of failure of equipment (part, component, subsystem) on crew safety or mission success. Criticality is non-numerical and is classified as follows:

Criticality I - Those items whose failure may result in loss of crew.

Criticality II - Those items whose failure may result in loss of mission.

Criticality III - Those items whose failure does not affect mission success or crew safety.

c. Production hardware - Hardware that is manufactured with the same tooling, processes, quality control procedures, and to the same design as that which will be used in manned flight.

d. Failure - The inability of a system, subsystem, component, or part to perform its required function within specified limits under specified conditions for a specified duration.

e. Operational cycle - The period of time extending from the beginning of acceptance tests to the end of a mission that a system, subsystem, component or part is expected to operate under sequential application of environments, induced or natural, including, but not limited to, end-item test time, acceptance time, checkout time, transportation and handling time, the planned mission time, and any critical abort or emergency conditions. The planning mission for qualification shall be 14 days duration. Any deviation as to mission duration shall be submitted to NASA for approval.

4.3.2 <u>Ground Rules</u>. -

a. The qualification program is limited to tests conducted on individual parts, components, subassemblies, assemblies, and subsystems. The qualification program shall consist of a series of tests at any or all assembly levels listed above and shall, in general, start with tests at the lower levels and proceed to levels of higher assembly.

b. Production hardware shall be used throughout.

c. Acceptance tests shall precede all qualification tests.

d. No refurbished equipment shall be used without specific NASA approval.

e. Functional operation is required. During all qualification tests all interfaces shall be present or simulated.

f. Adjustments will be permitted during an operational cycle only if they are part of a normal procedure.

g. Limited life items and single shot devices may be replaced at the completion of satisfactory operation through their life time requirement.

h. Any failure shall be cause for positive correction action and shall be cause for complete restart of that test series. In event of failure, the contractor shall immediately advise NASA.

i. Requalification shall be performed when:

   (1) Design or manufacturing processes are changed to the extent that the original tests are invalidated.

   (2) Inspection, test, or other data indicate that a more severe environment or operational condition exists than that to which the equipment was originally qualified.

   (3) Manufacturing source is changed.

j. Qualification by similarity may be accepted provided:

   (1) The item was qualified to the Apollo environmental levels, and,

   (2) The item was fabricated by the same manufacturer with the same processes and quality control.

   (3) The item was designed to equivalent specifications required of the Apollo designs.

k. Qualification testing shall include both natural and induced environments which simulate, as closely as required, the anticipated environments during the operational cycles in level, range, and sequence. Combined environments shall be used when necessary and practical.

l. Where redundancy in design exists, the qualification test program will assure that each redundant component will be included in the test program.

m. Qualification test specifications shall be written for each item and the qualification test program will fully encompass the design specification requirements.

n. As a general rule, it is not economically practical or feasible to conduct qualification tests on complete subsystems. Accordingly, most of the qualification tests should be conducted on lower levels of assemblies to the degree necessary to provide confidence on a subsystem basis. This will be done by conducting tests at each hardware level such that when the total qualification program on a subsystem is completed all items of hardware and all operational modes will, as a minimum, be tested an amount equivalent to a subsystem qualification test. This is commonly called an "equivalent" subsystem.

o. The qualification program shall be established in two phases, namely, that required to support Block I vehicle missions and secondly, Block II LOR missions. The program should be established so that hardware required for qualification is selected on a time phase basis throughout the production program as an objective. The number of units required prior to the first manned Block I flight should be reduced to a minimum. In determining the number of units required for qualification, all prior development tests including integrated ground tests should be considered in determining the number of units required. Portions of the development tests may be used to reduce the qualification test program provided all qualification requirements are met and prior NASA approval is obtained.

p. Qualification tests supporting a particular vehicle shall be completed prior to that vehicle being delivered from the contractor's plant. As a minimum the following qualification will be completed before the first Saturn 1B flights. One set of equipments will be subjected to sequential, singly applied environments at design limit conditions. Another set is required to complete two operational cycles at nominal mission conditions, the first cycle without failure.

q. Subsequent to the completion of the qualification test program further tests shall be conducted at conditions more severe than design-limit. The purpose of these tests shall be to determine failure modes actual design margins.

4.3.3 S/C Development Test. - The S/C development tests shall be in accordance with the Apollo Spacecraft Development Test Plan 62-109, Volume V.

4.4 Configuration Management Provisions.

4.4.1 Change Control. - NAA/S&ID shall maintain an effective configuration control program to control the incorporation of engineering changes affecting engineering orders and drawings, specifications, procurement documents, quality control, inspection and test procedures, process, manufacturing, and operation instructions, and similar documents.

5. PREPARATION FOR DELIVERY

5.1 Preservation, Packaging, and Packing. - Preservation, packaging, and packing shall be in accordance with NAA/S&ID procedures, provided the procedure assures adequate protection in accordance with delivery modes, destinations, and anticipated storage periods.

5.2 Handling. - Handling shall be in accordance with NAA/S&ID procedures.

6. NOTES

6.1 Reference Axes. - The reference axes of the CSM shall be orthorgonal and shall be identified as shown in Table I.

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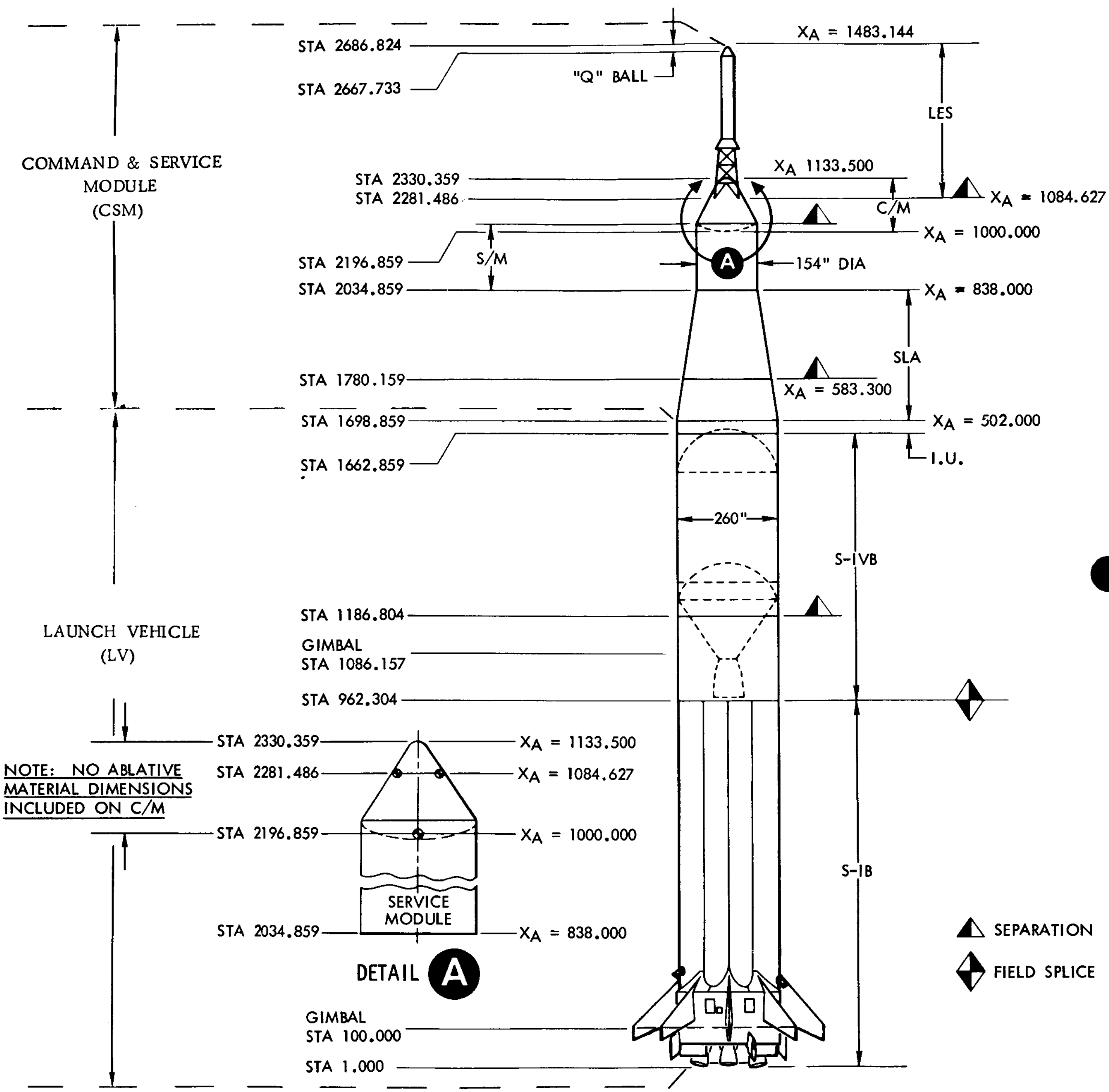

Figure 1. Saturn C-IB Configuration

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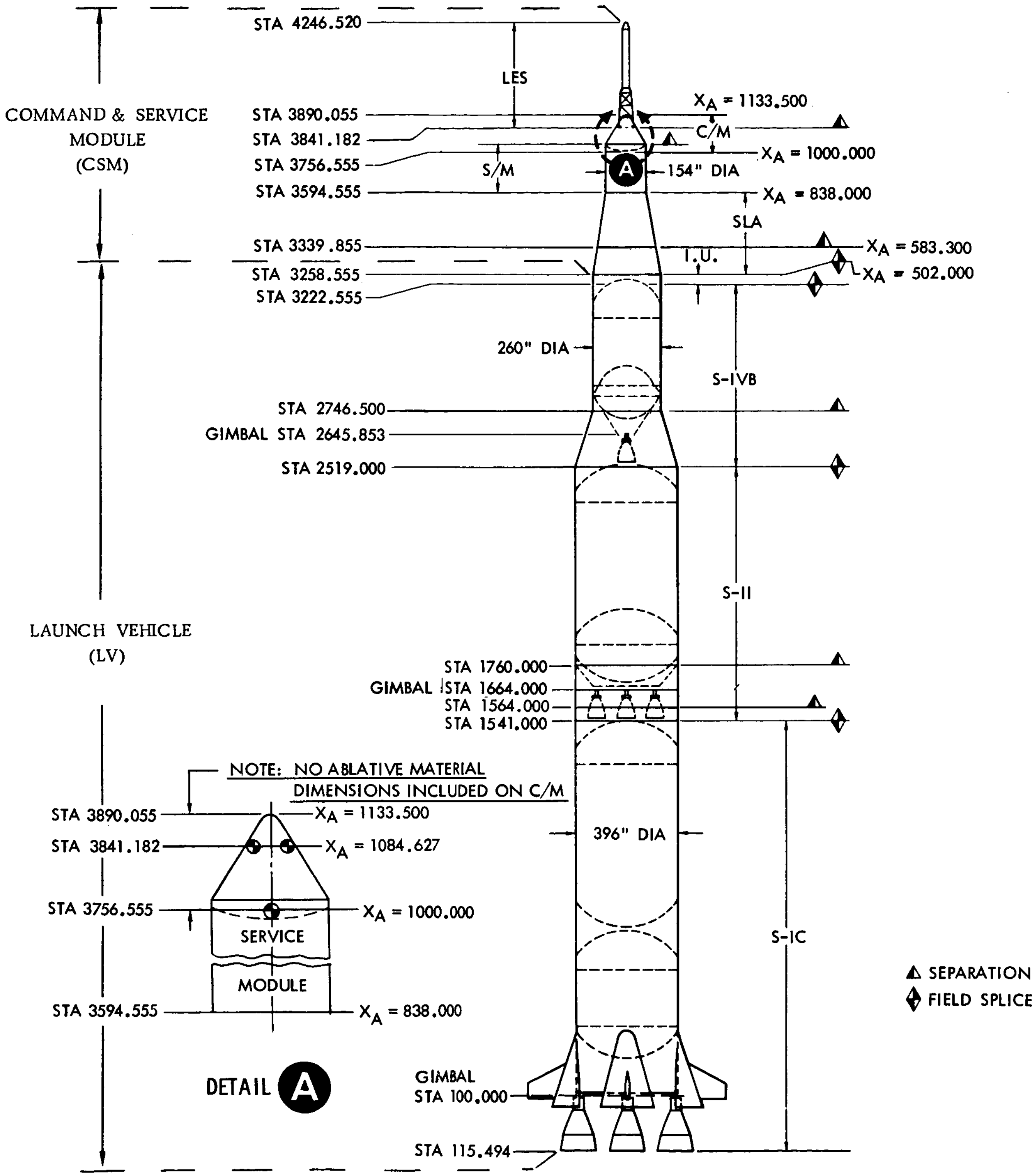

Figure 2. Saturn V LOR Configuration

[redacted]

[redacted]

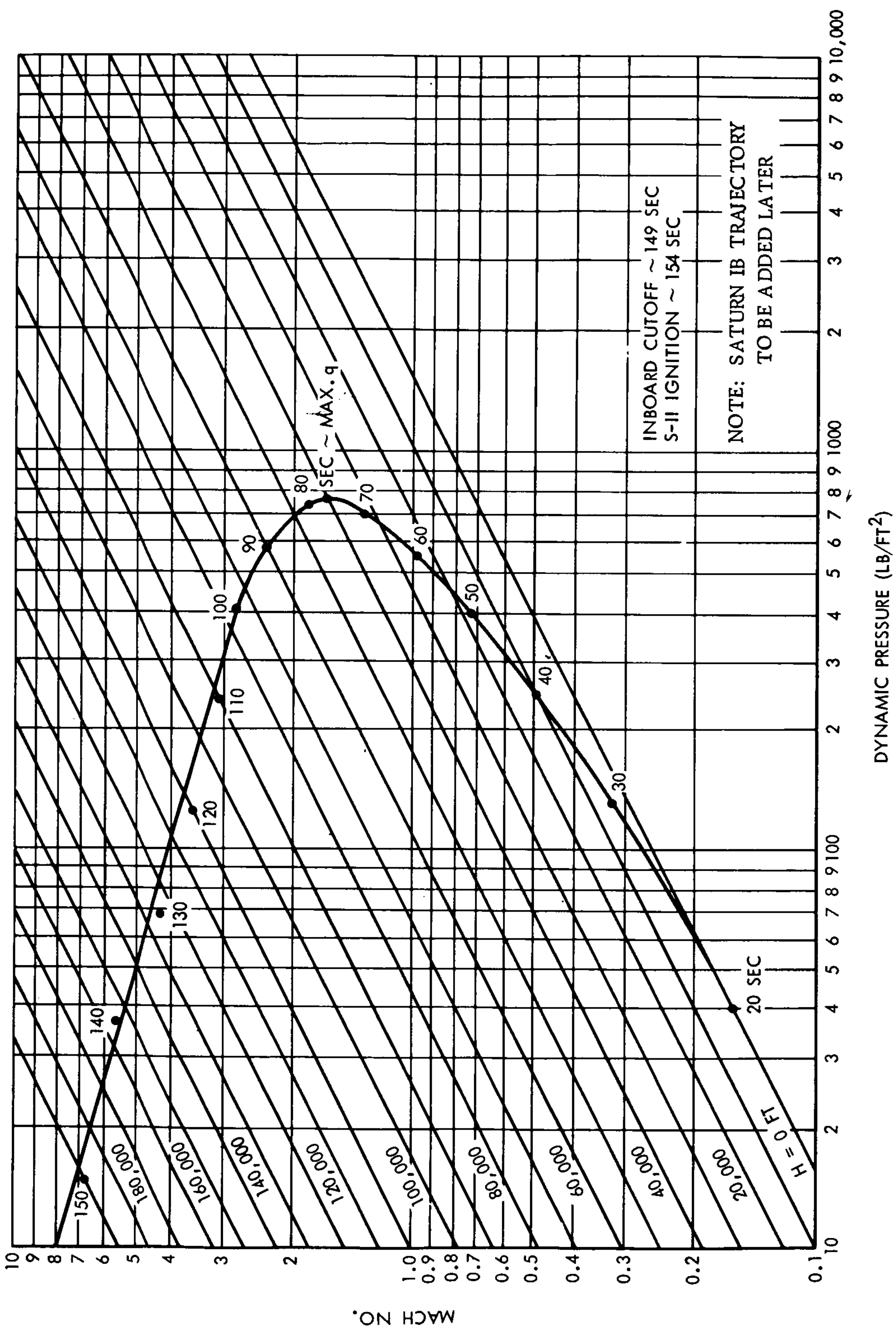

Figure 3. Saturn V Two-Stage Boost Trajectory - 100 NM Orbit

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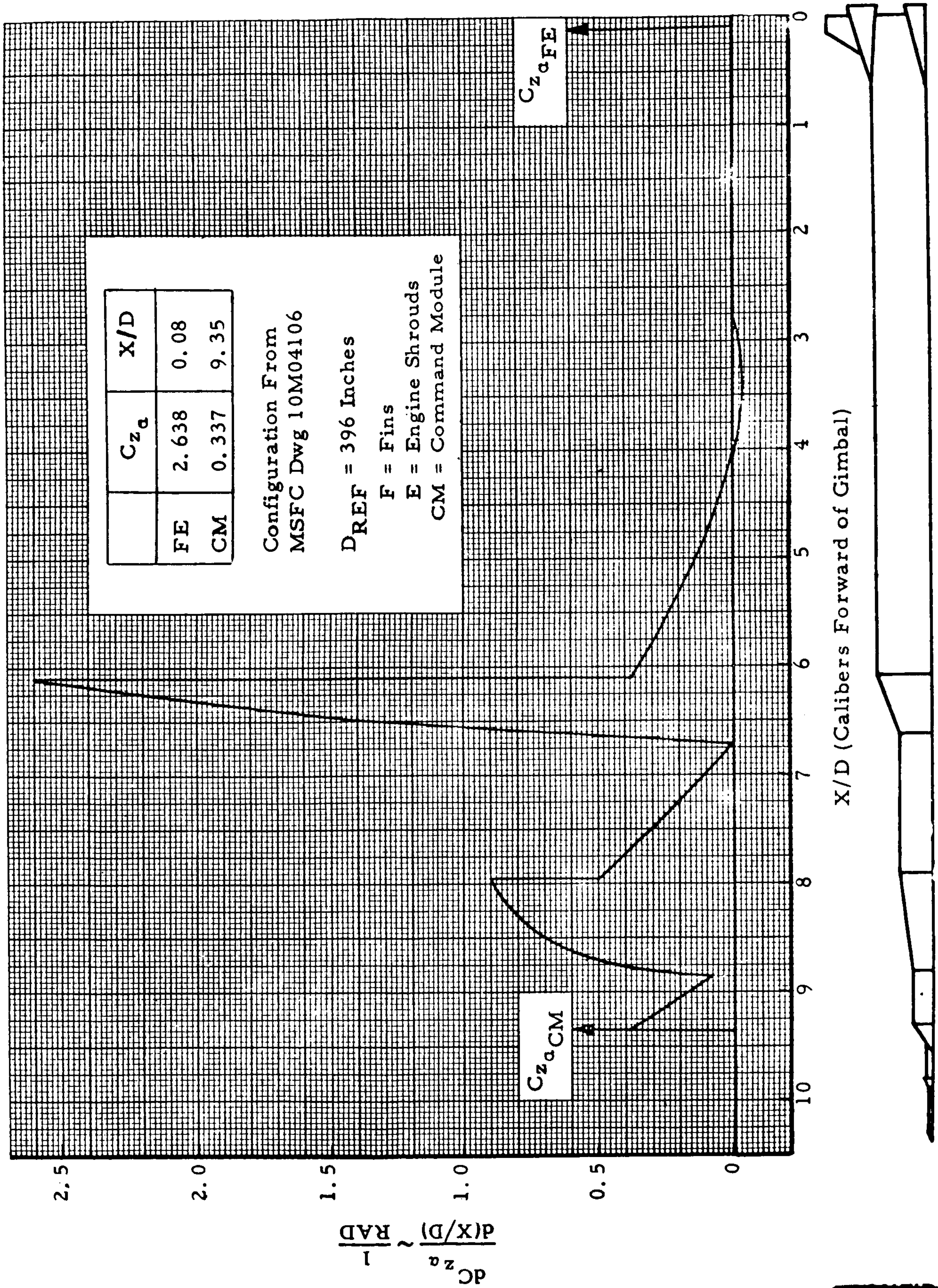

Figure 4. Linear Load Distribution for Mach Numbers 1.35 (Saturn V)

[redacted]

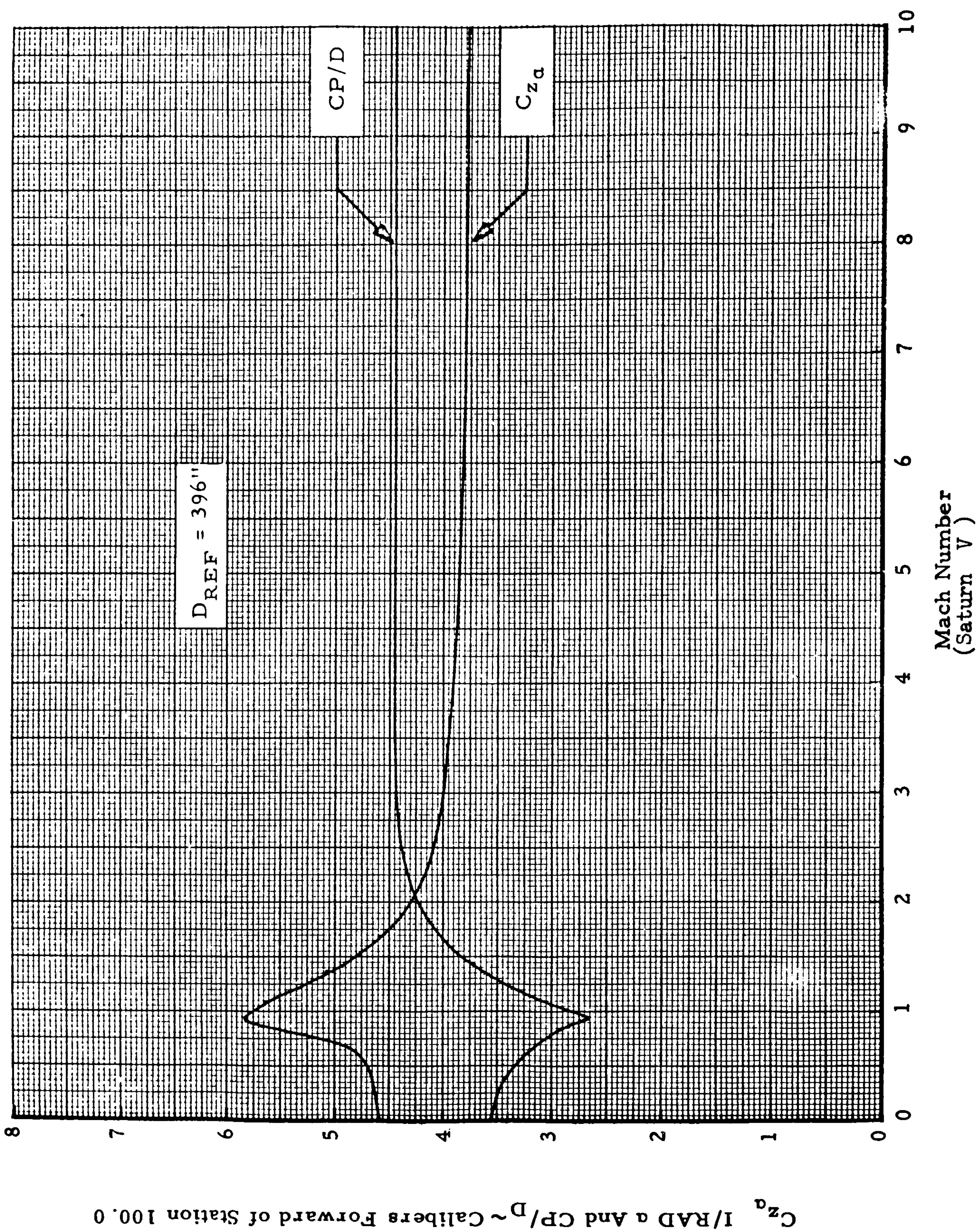

Figure 5. Gradient of Normal Force Coefficient and Center of Pressure Versus Mach Numbers

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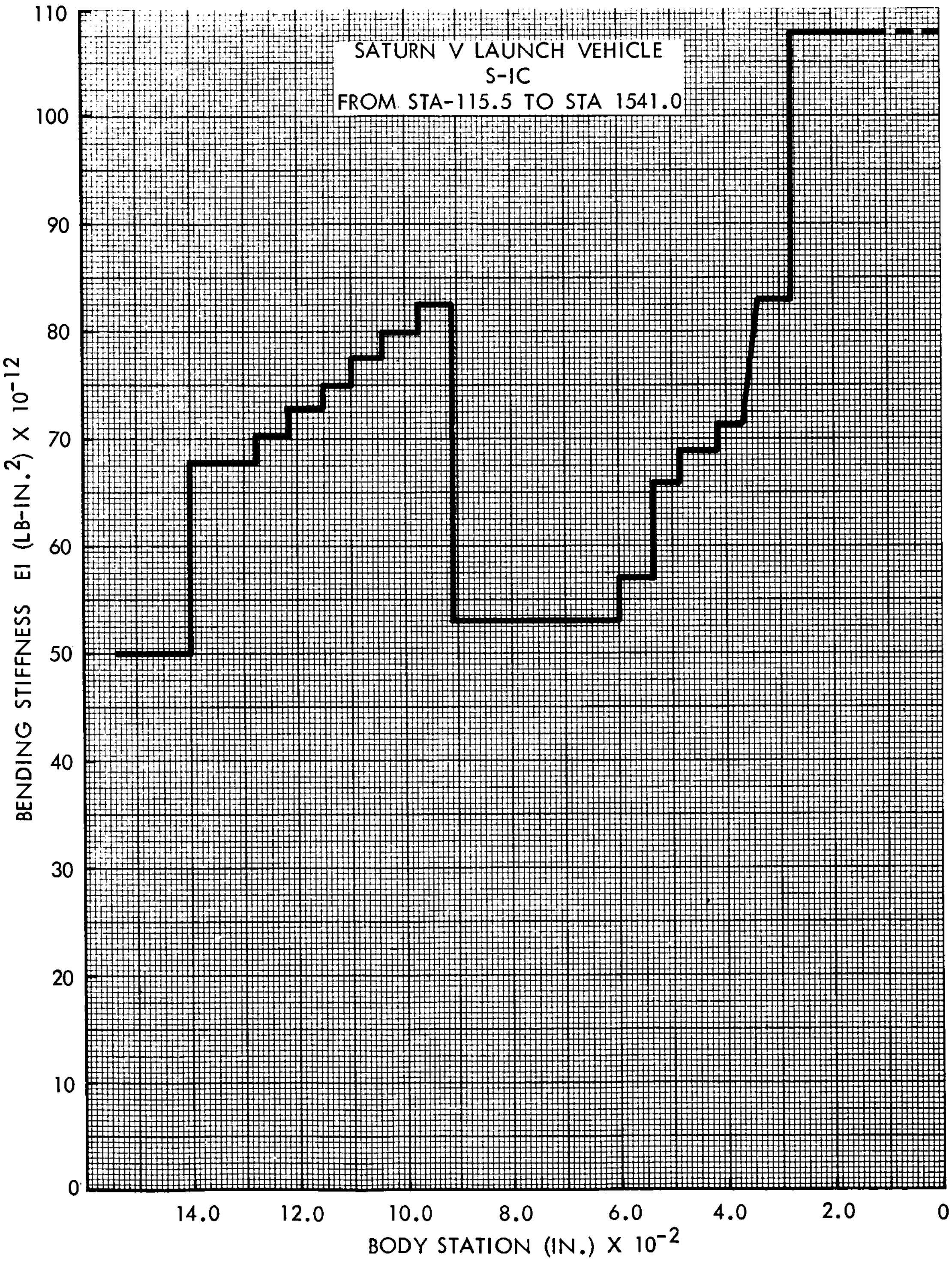

Figure 6. Saturn V, S-IC, EI Distribution

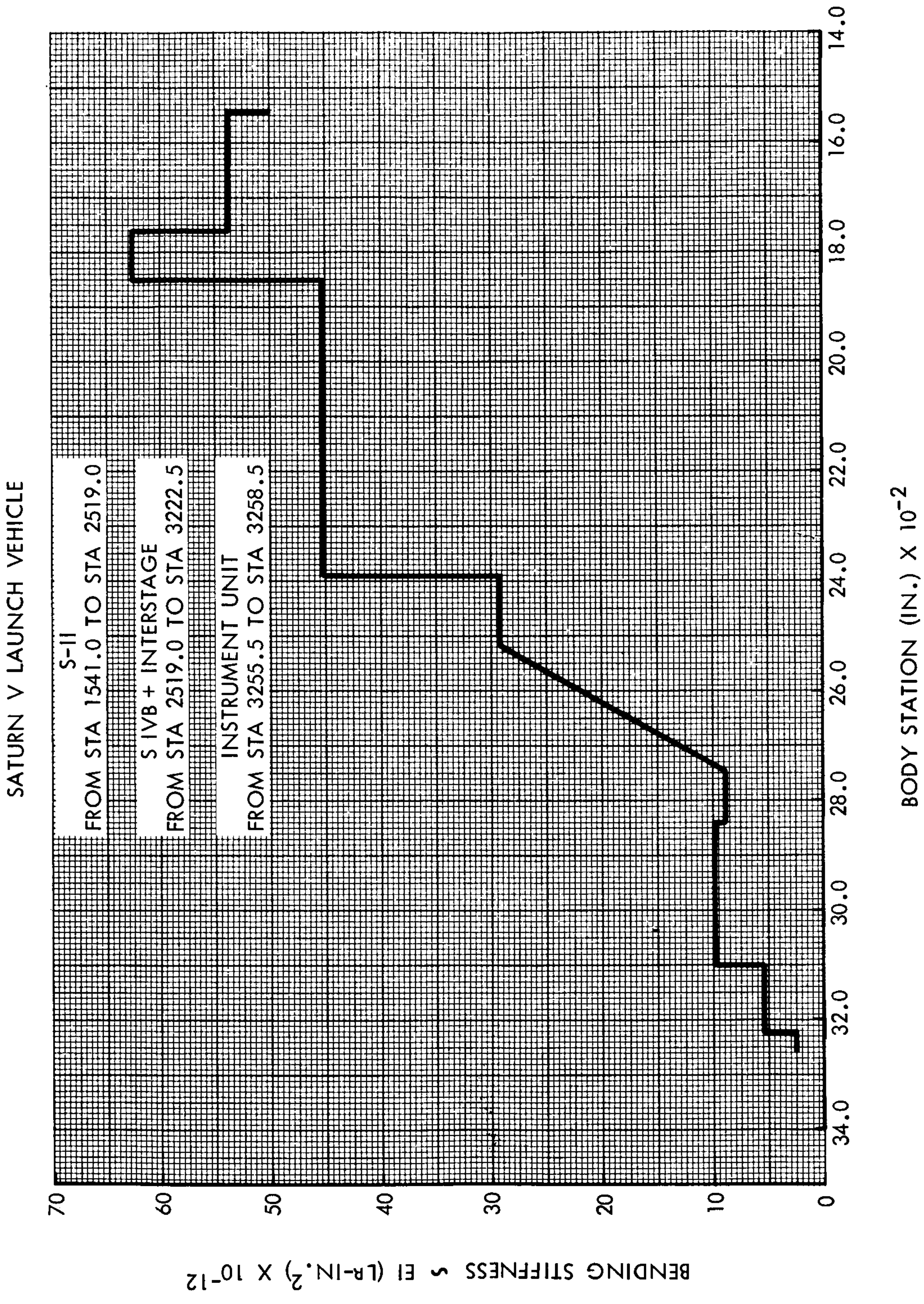

Figure 7. Saturn V, SSI, EI Distribution

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SATURN V LAUNCH VEHICLE

S-IC
FROM STA-115.5 TO STA 1541.0

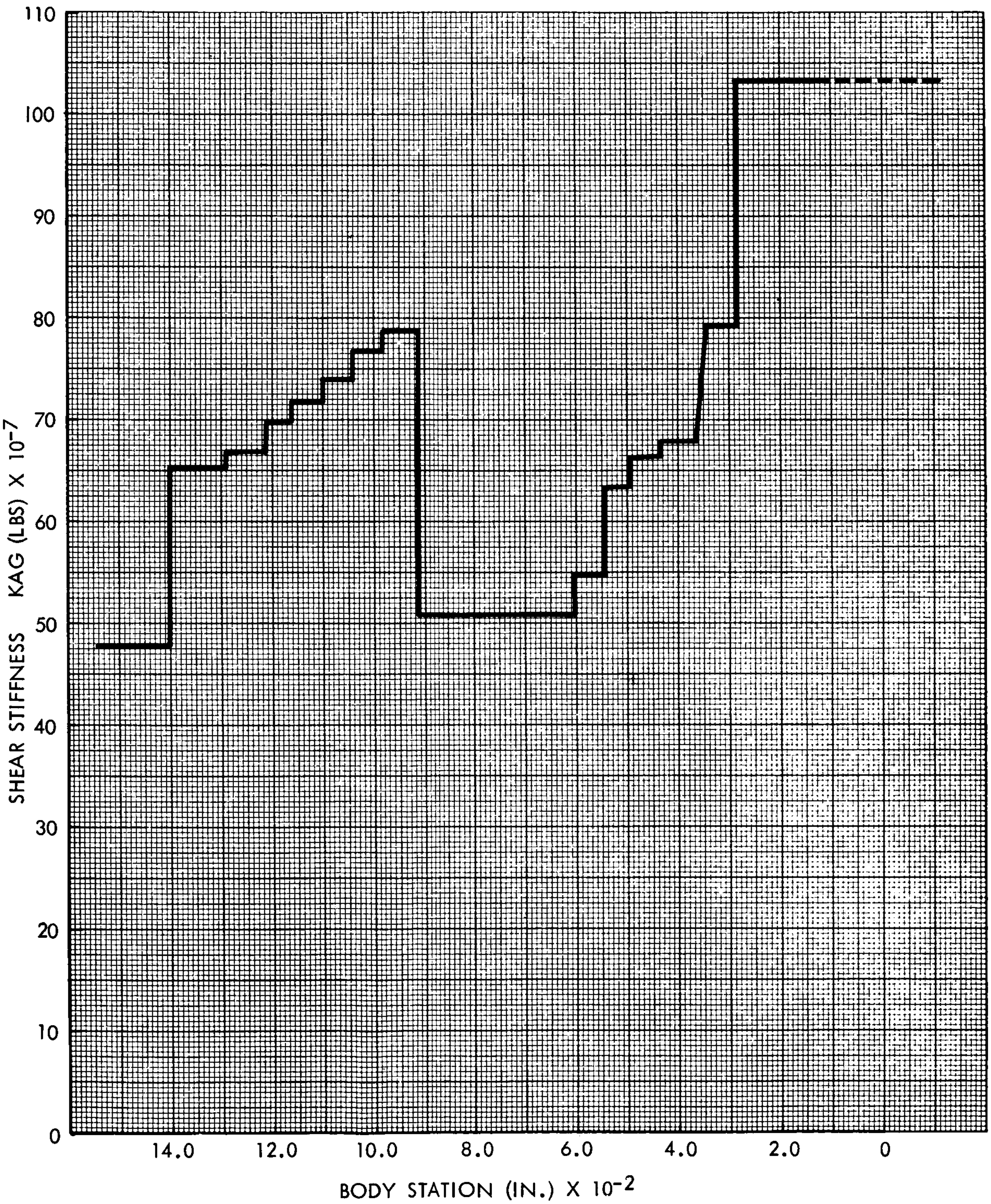

Figure 8. Saturn V, S-IC, KAG Distribution

[redacted]

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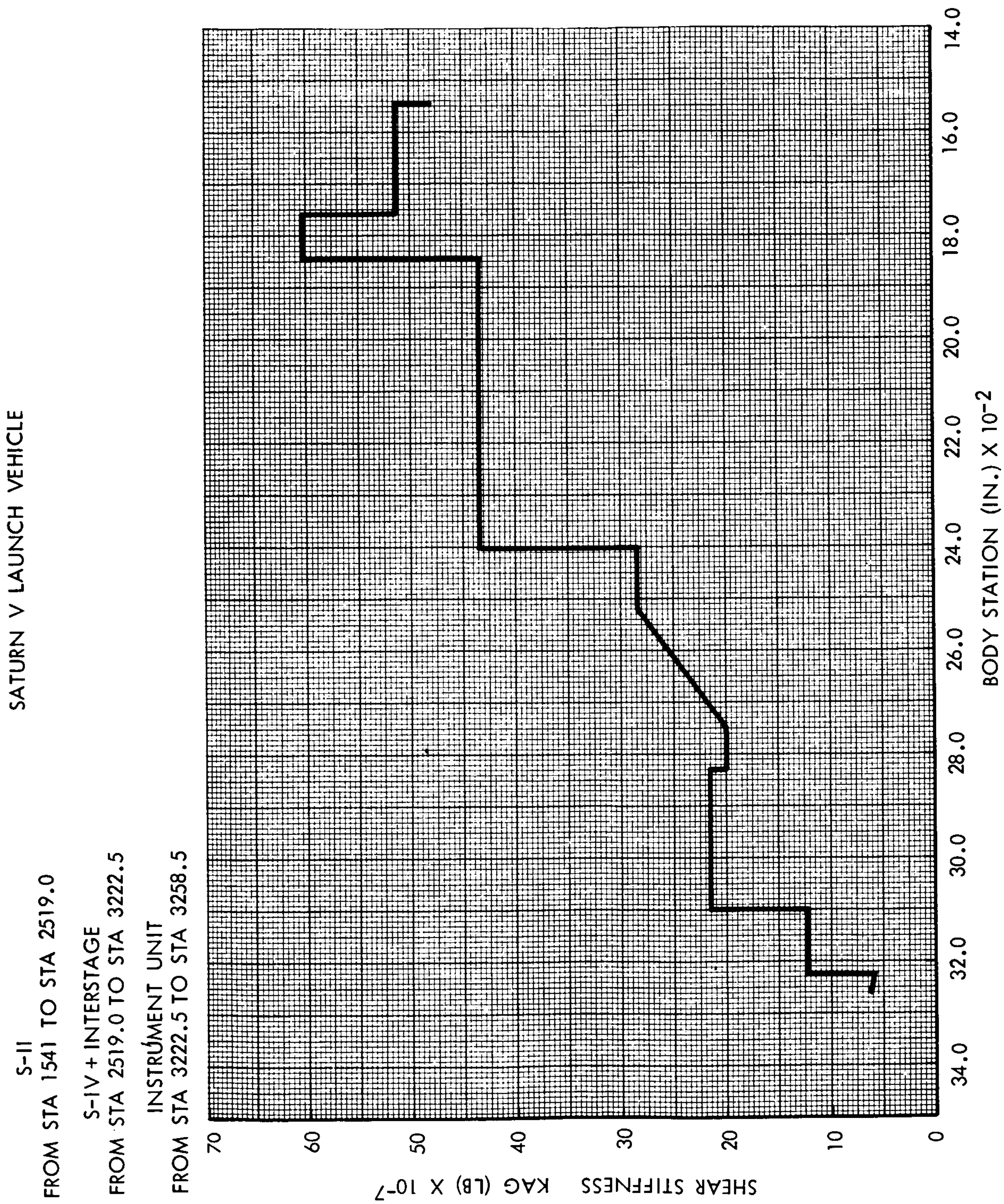

Figure 9. Saturn V, S-II, KAG Distribution

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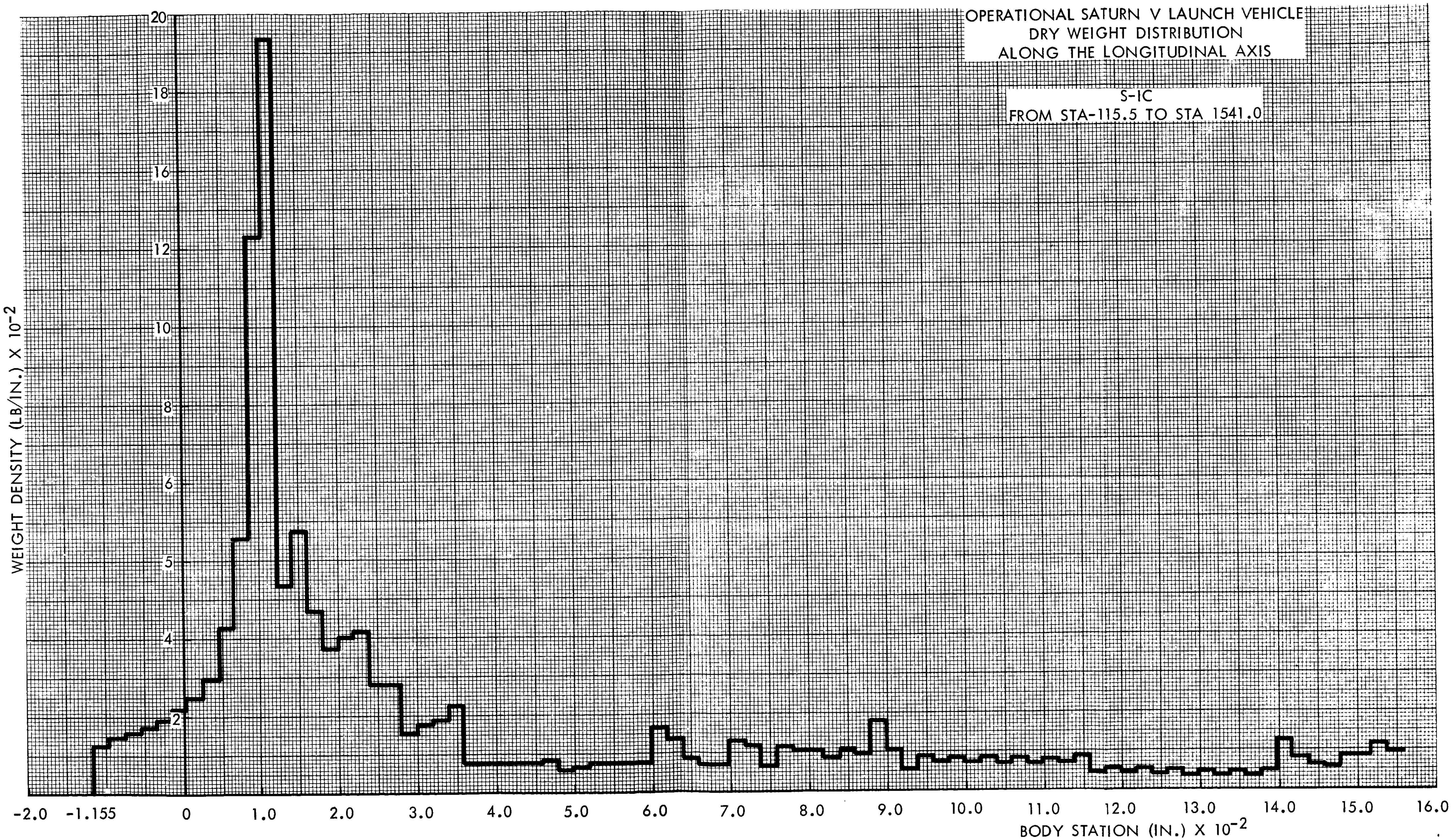

Figure 10. Saturn V, S-IC, Dry Weight Distribution

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OPERATIONAL SATURN V LAUNCH VEHICLE
WET WEIGHT DISTRIBUTION
ALONG THE LONGITUDINAL AXIS
FROM STA-115.5 TO STA 1541

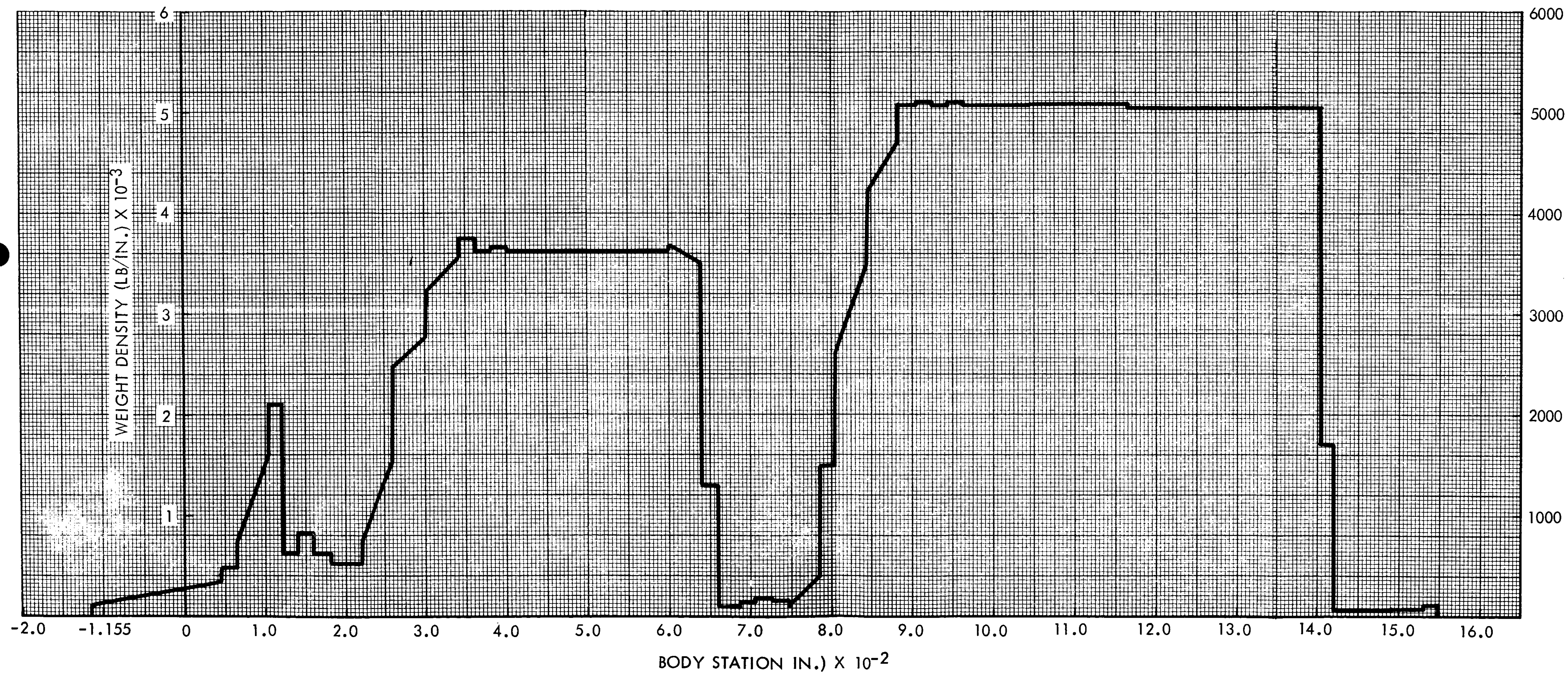

Figure 11. Saturn V, S-IC, Wet Weight Distribution

OPERATIONAL SATURN V LAUNCH VEHICLE
FUEL WEIGHT DISTRIBUTION
ALONG THE LONGITUDINAL AXIS

S-IC
FROM STA-115.5 TO STA 1541.0

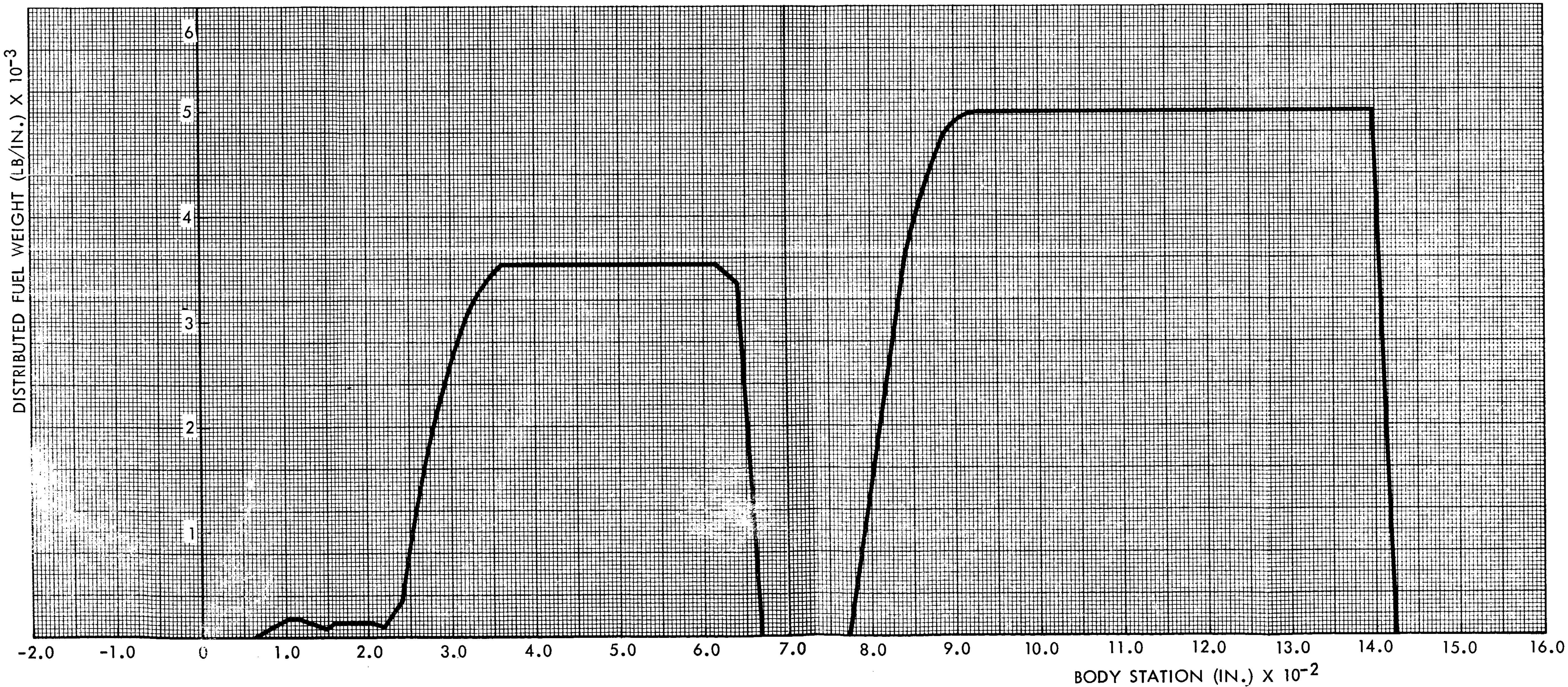

Figure 12. Saturn V, S-IC, Full Weight Distribution

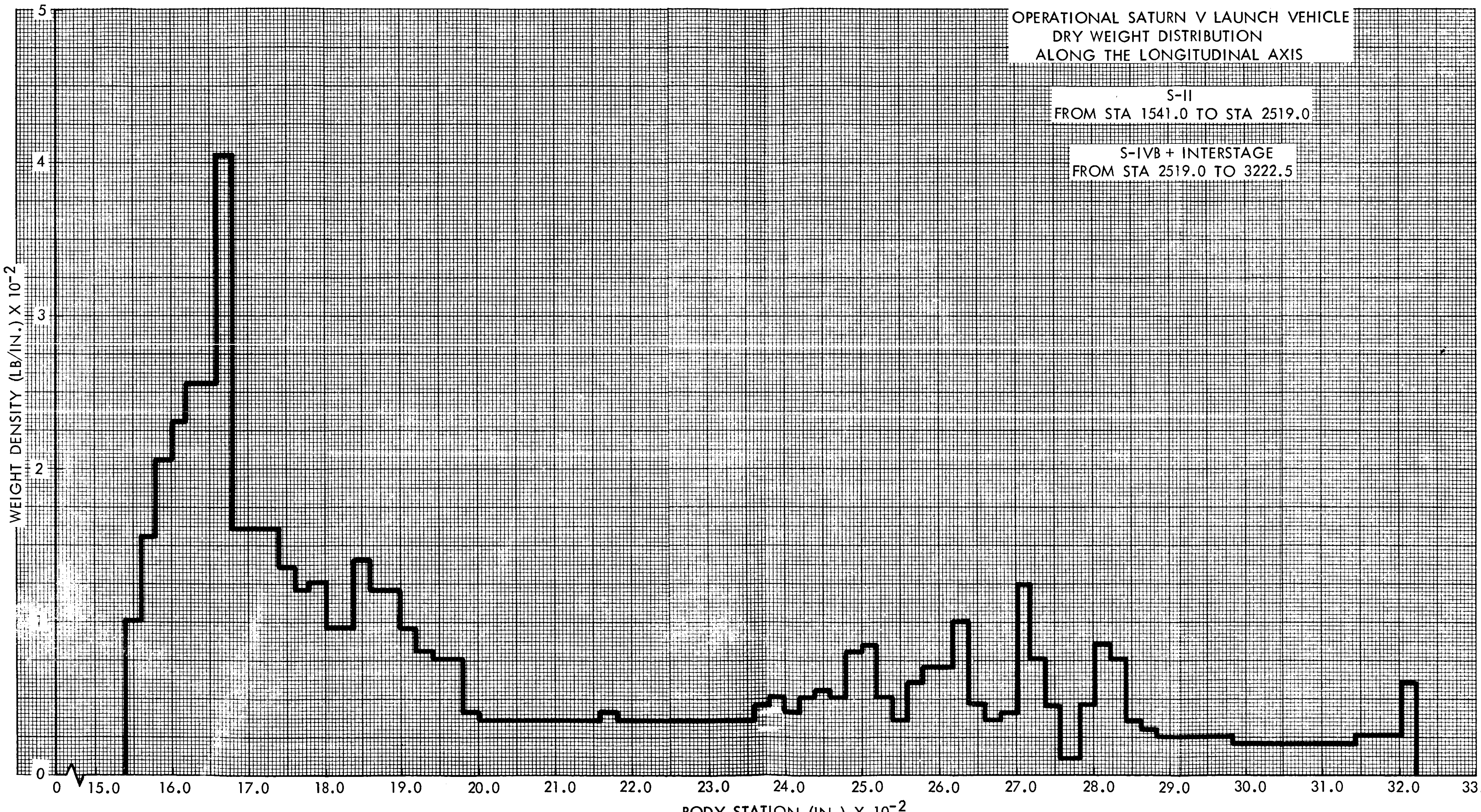

Figure 13. Saturn V, S-II, Dry Weight Distribution

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OPERATIONAL SATURN V LAUNCH VEHICLE
WET WEIGHT DISTRIBUTION
ALONG THE LONGITUDINAL AXIS

S-II
FROM STA 1541.0 TO STA 2519.0

S-IVB + INTERSTAGE
FROM STA 2519.0 TO STA 3222.5

INSTRUMENT UNIT
FROM STA 3222.5 TO STA 3258.5

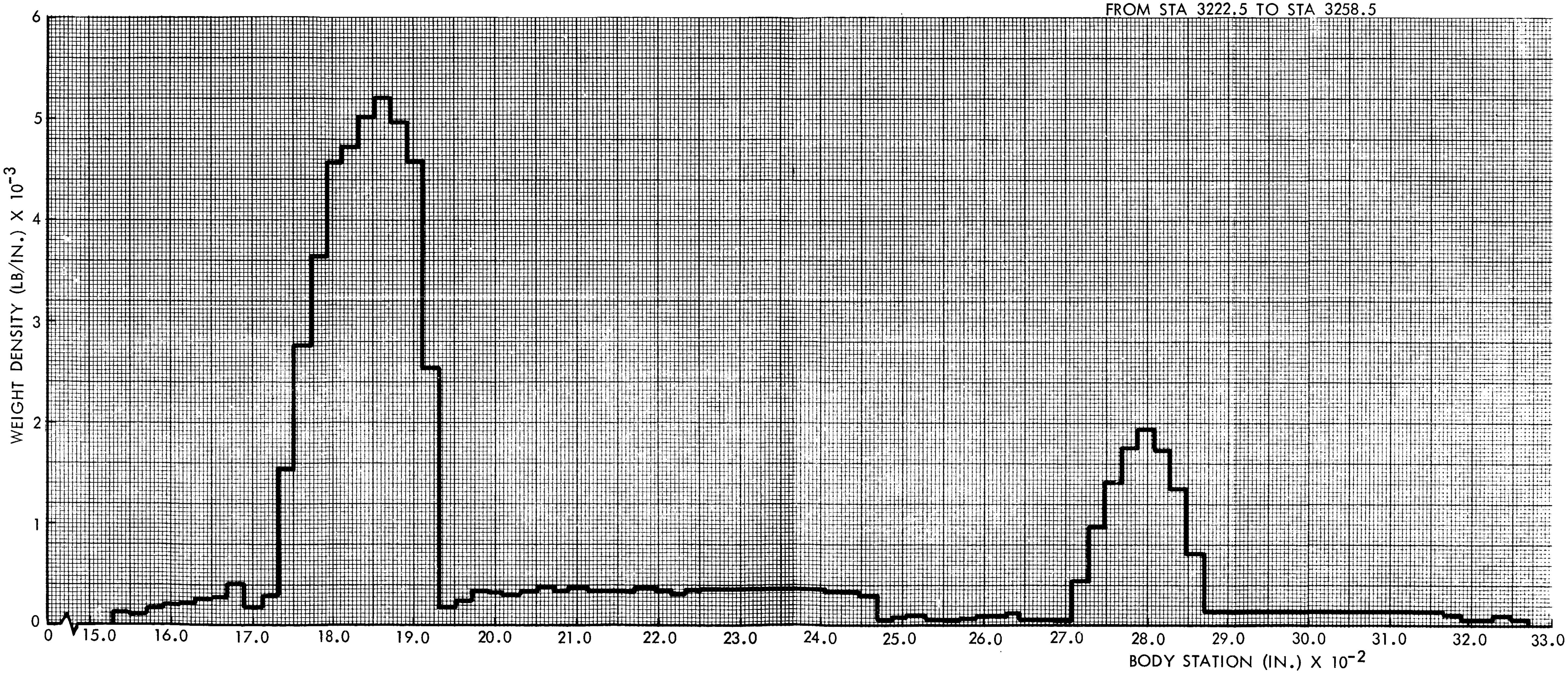

Figure 14. Saturn V, S-II, Wet Weight Distribution

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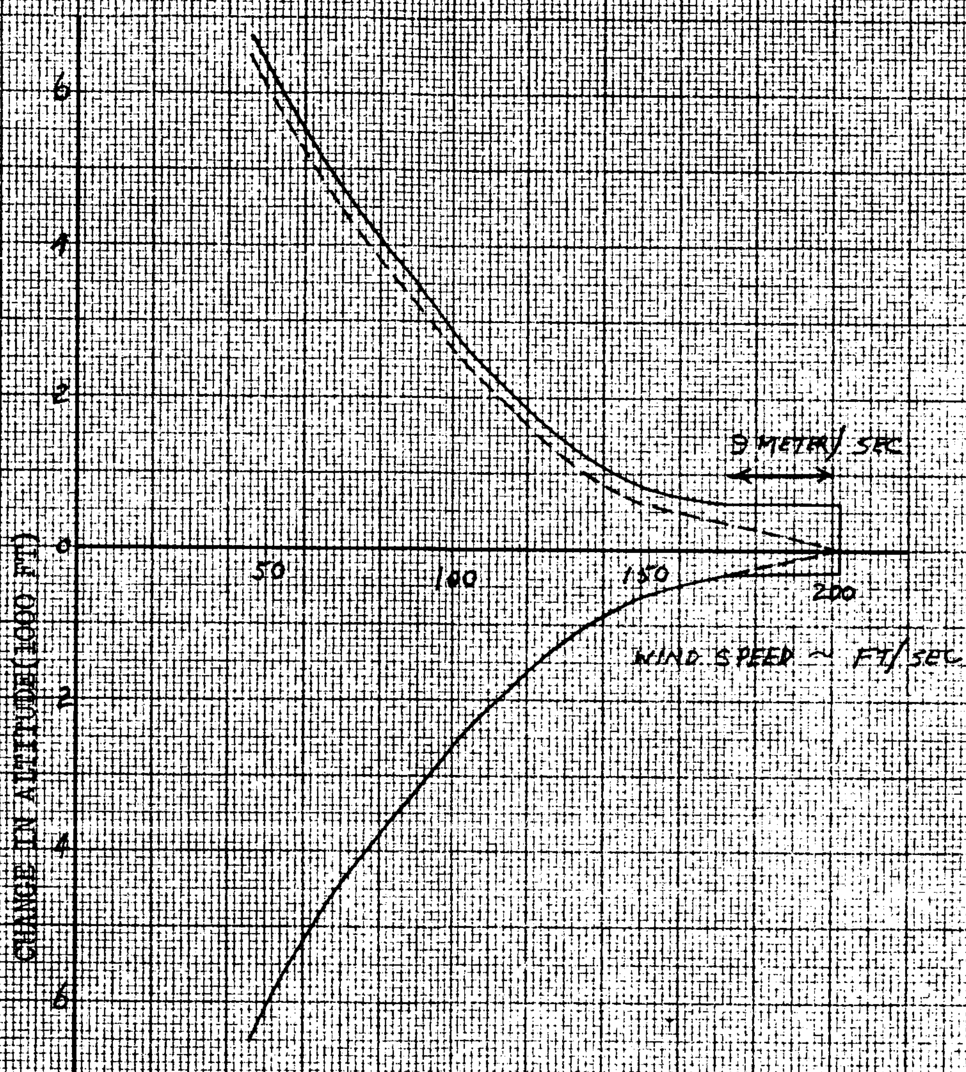

WIND SHEAR PROFILE
(88% WIND SPEED)

NOTE: THIS DATA IS APPLICABLE ONLY IN THE ALTITUDE RANGE FROM 32800 FEET TO 46000 FEET

Figure 15. Wind Shear Profile

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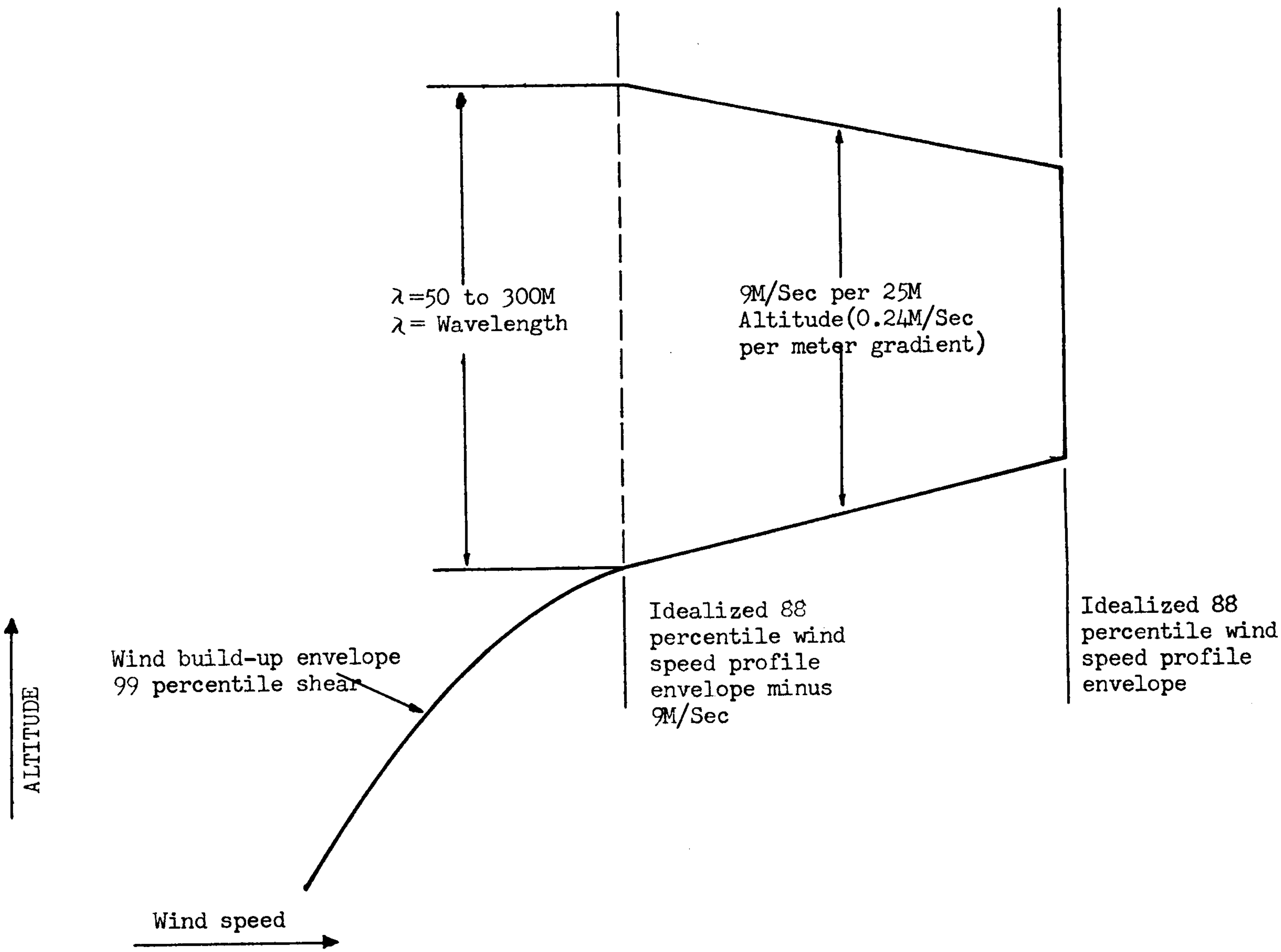

Figure 16. Relationship Between Established Gust Characteristics (Quasi Square Wave Shape) and the Idealized Windspeed (Quasi Steady State) Profile Envelope

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VIBRATION TIME HISTORY
ATMOSPHERIC FLIGHT

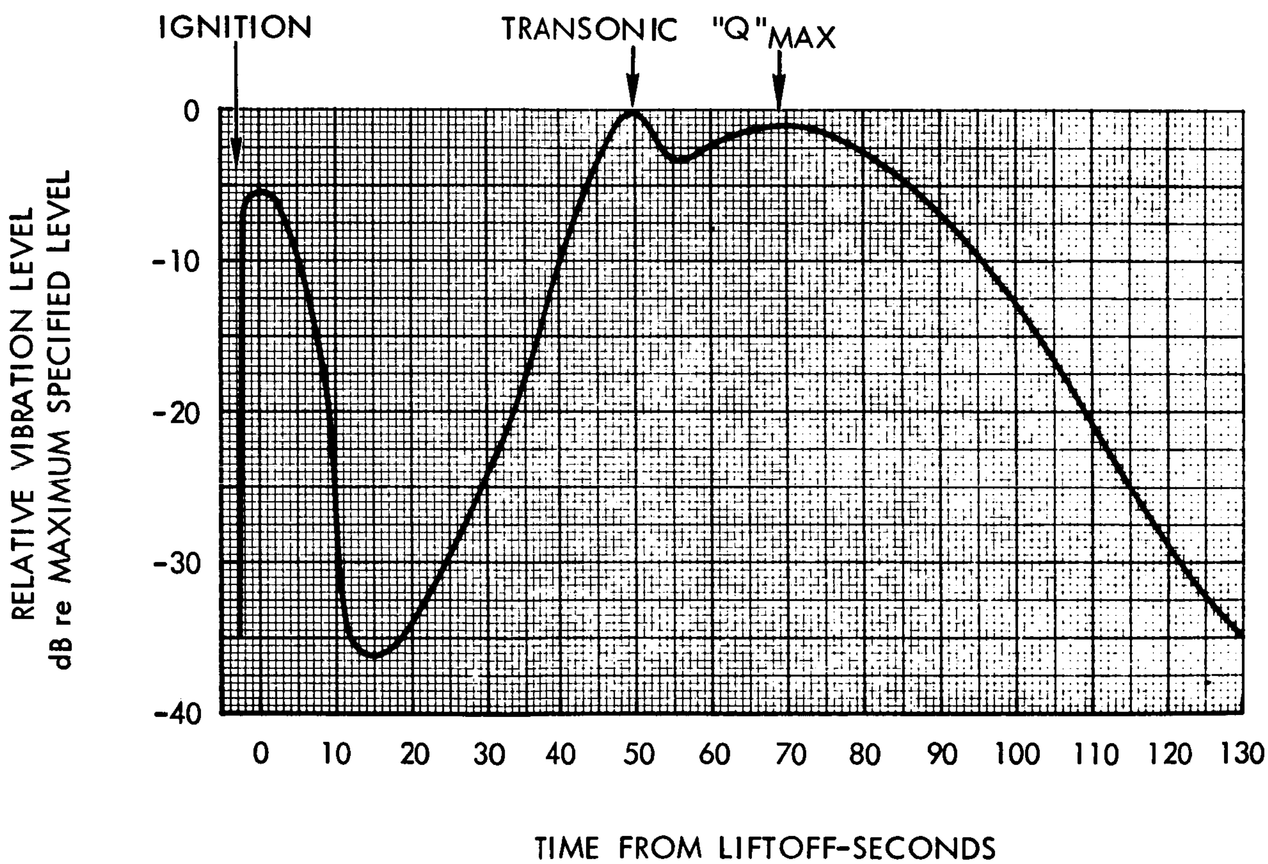

Figure 17. Vibration Time History - Atmospheric Flight

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VIBRATION
LAUNCH ESCAPE SYSTEM
ATMOSPHERIC FLIGHT

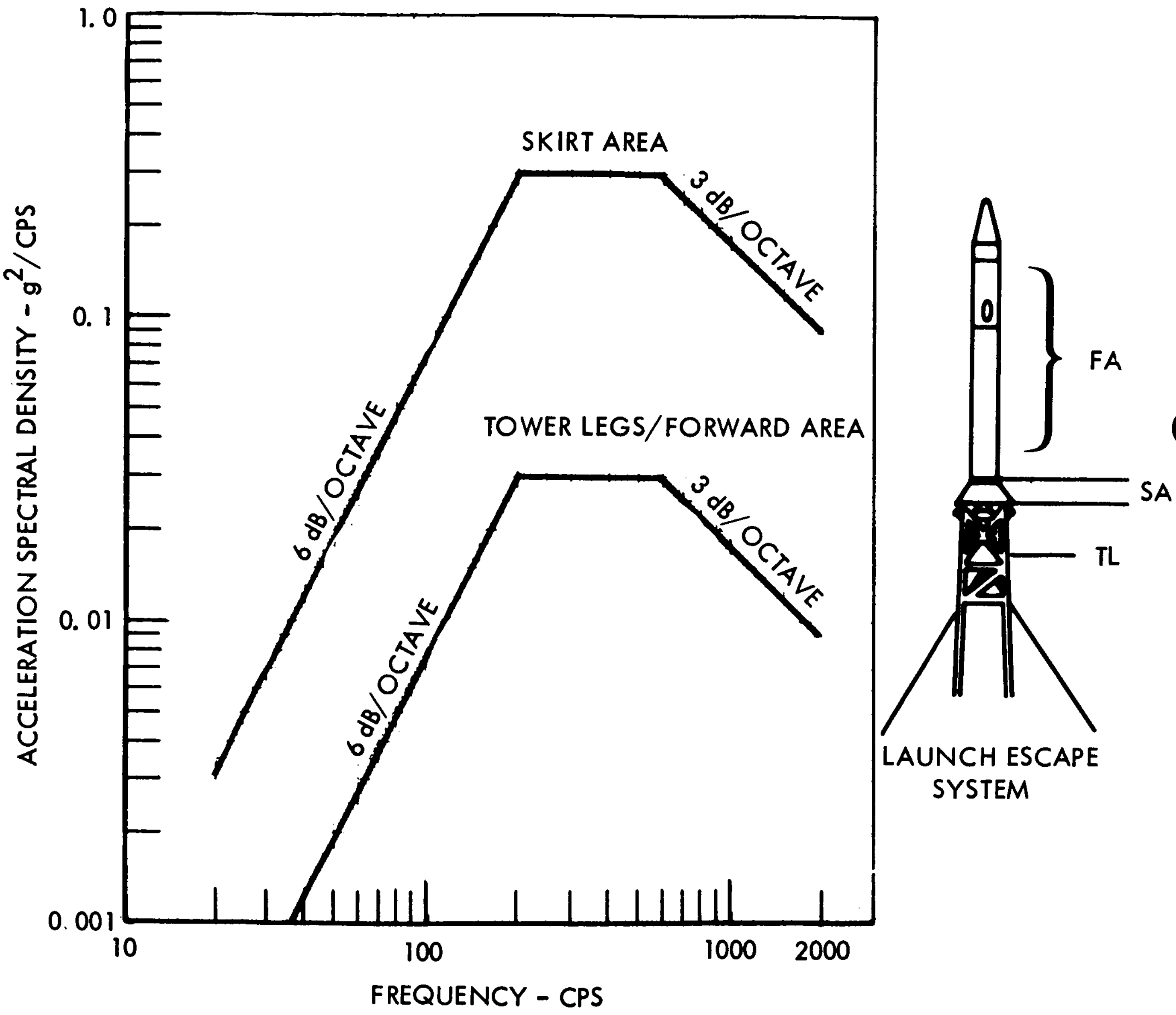

Figure 18. Vibration LES - Atmospheric Flight

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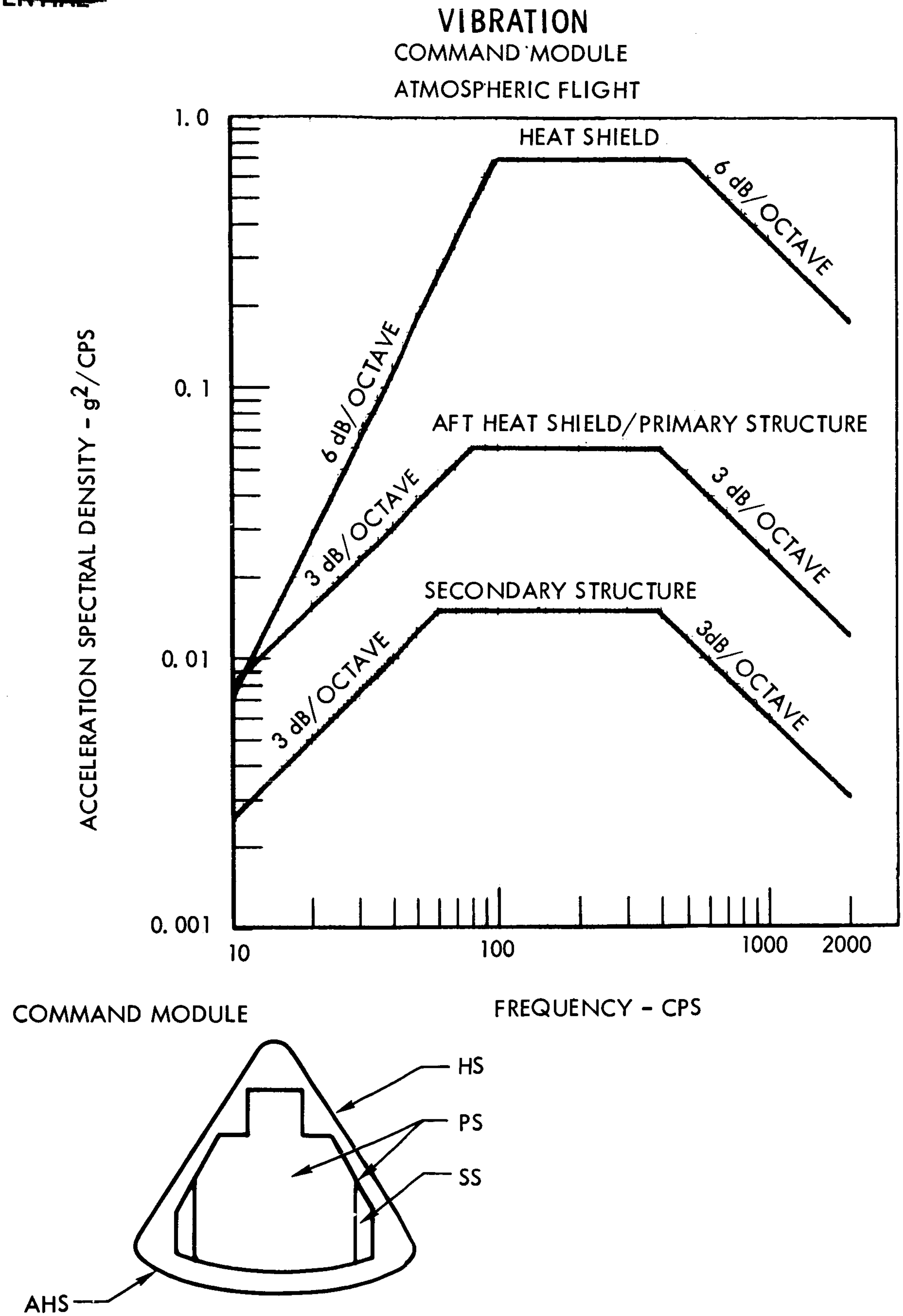

Figure 19. Vibration CM - Atmospheric Flight

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VIBRATION
SERVICE MODULE
ATMOSPHERIC FLIGHT

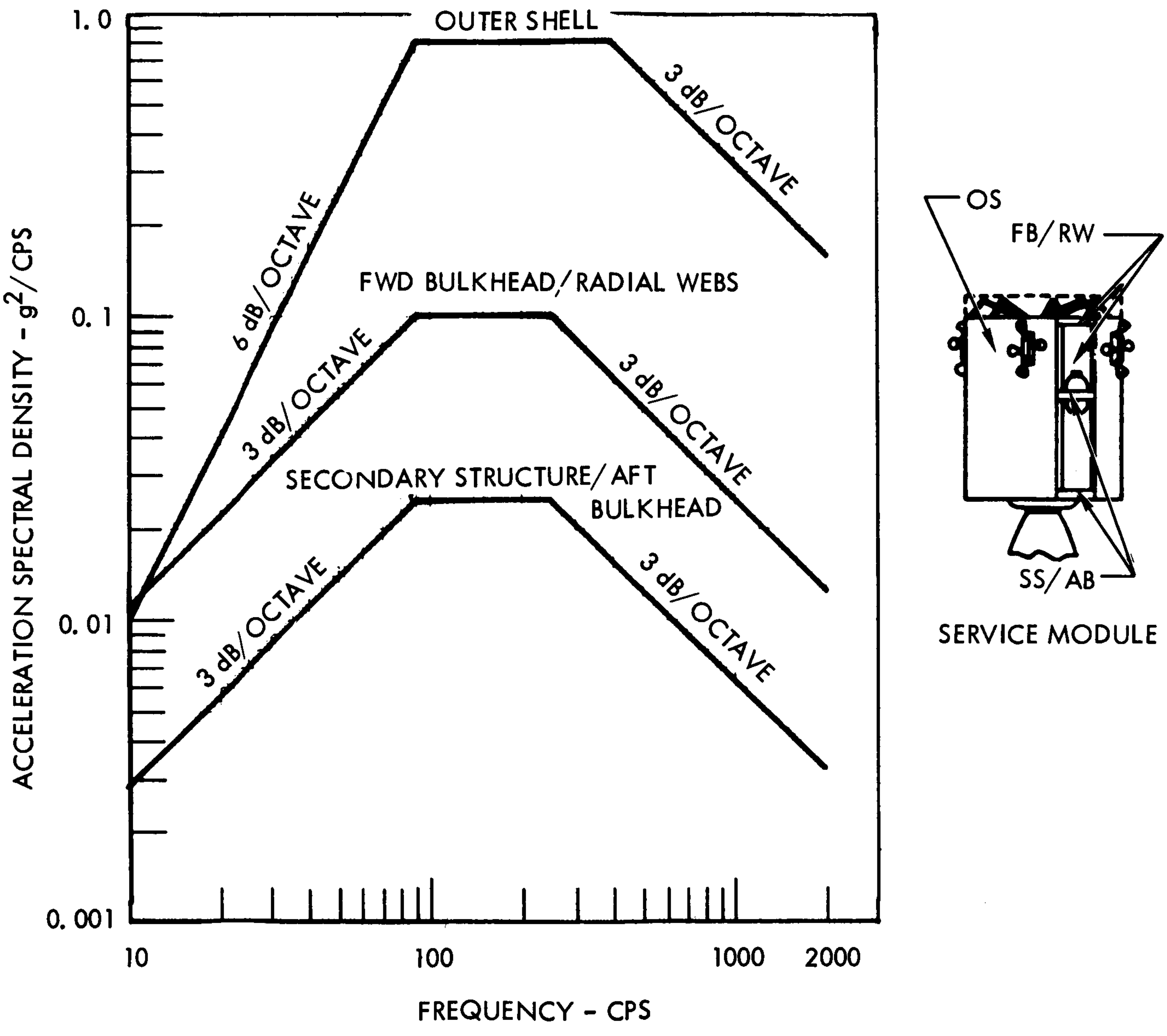

Figure 20. Vibration SM - Atmospheric Flight

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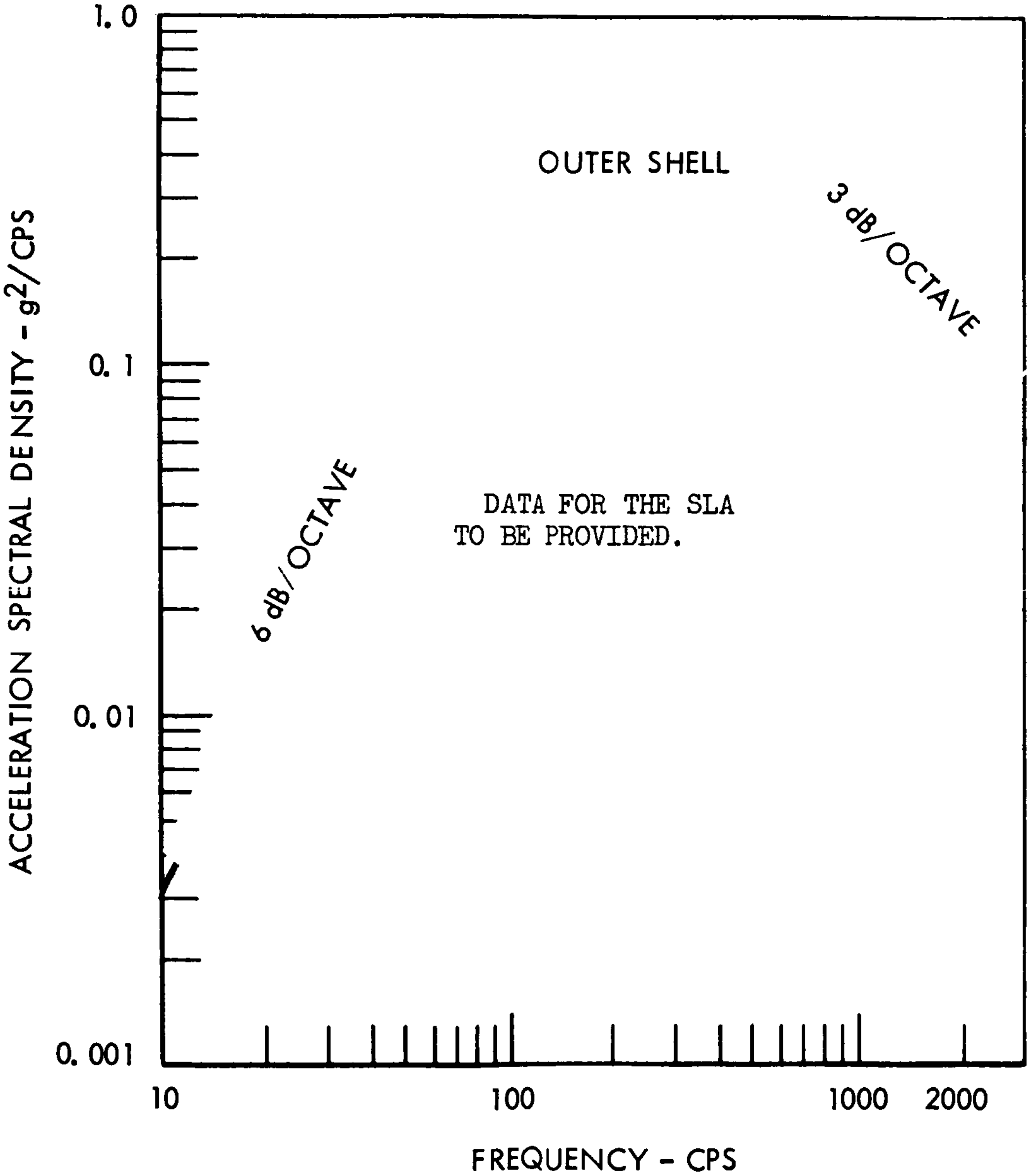
VIBRATION
SERVICE MODULE ADAPTER
ATMOSPHERIC FLIGHT



SM ADAPTER

Figure 21. Vibration SM Adapter - Atmospheric Flight

ACOUSTICS

LAUNCH ESCAPE SYSTEM
ATMOSPHERIC FLIGHT

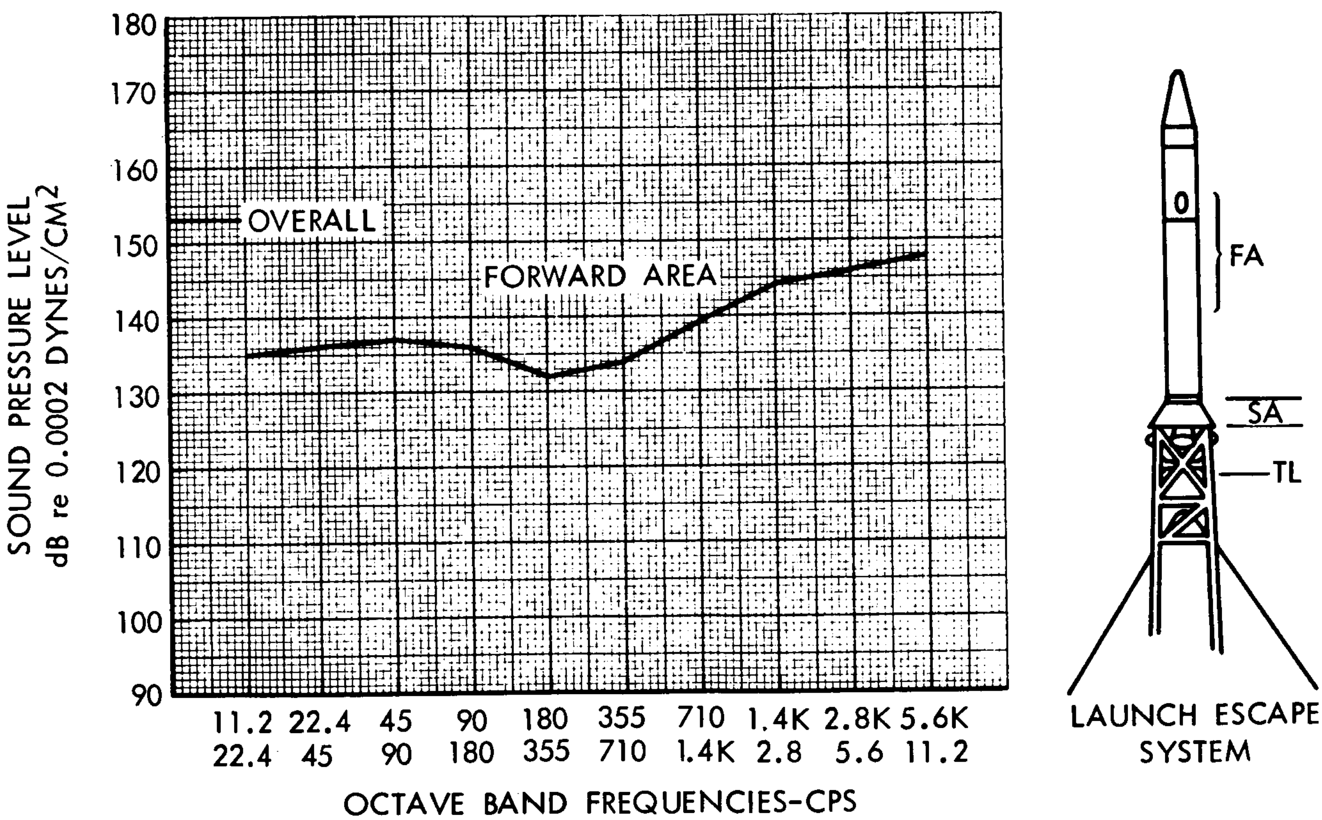


Figure 22. Acoustics LES - Atmospheric Flight - Forward Area

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ACOUSTICS

LAUNCH ESCAPE SYSTEM
ATMOSPHERIC FLIGHT

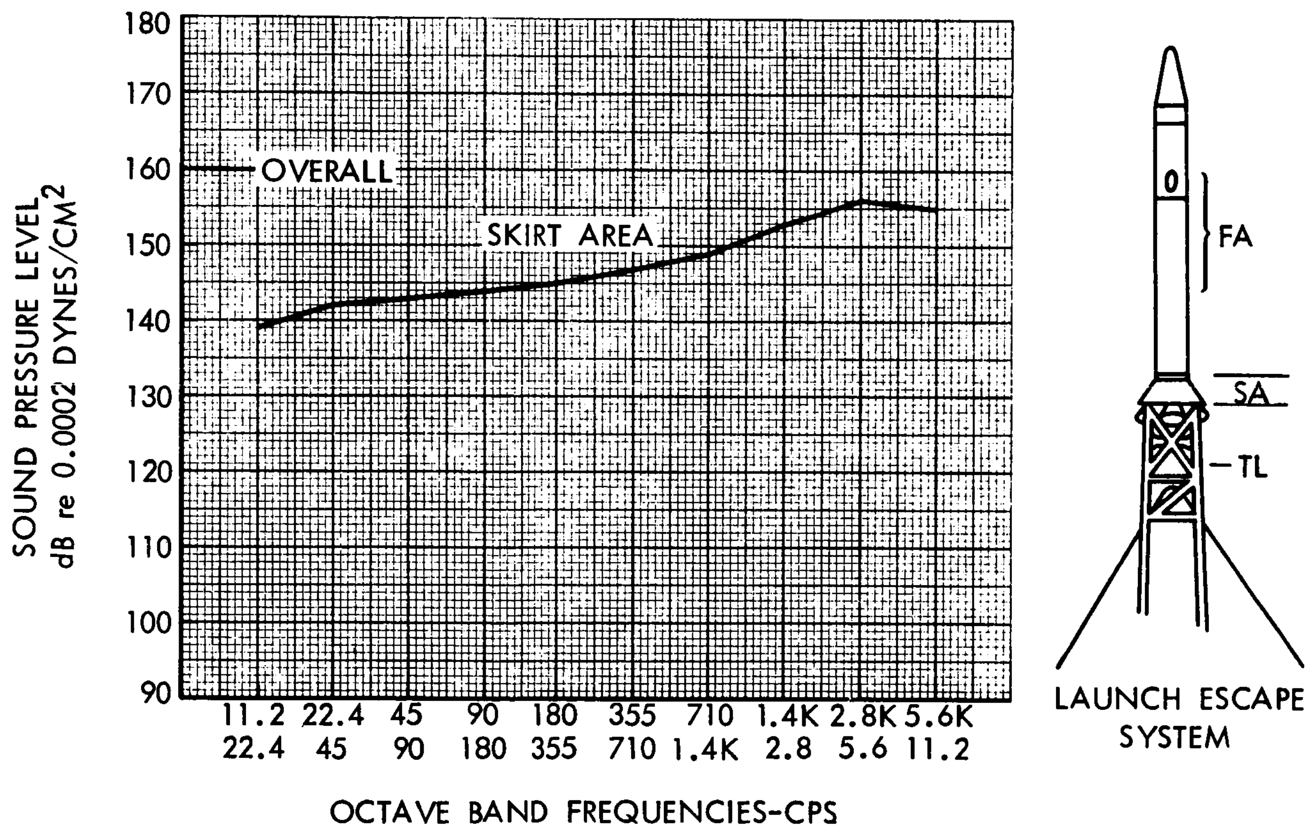

Figure 23. Acoustics LES - Atmospheric Flight - Skirt Area

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ACOUSTICS

COMMAND MODULE

ATMOSPHERIC FLIGHT

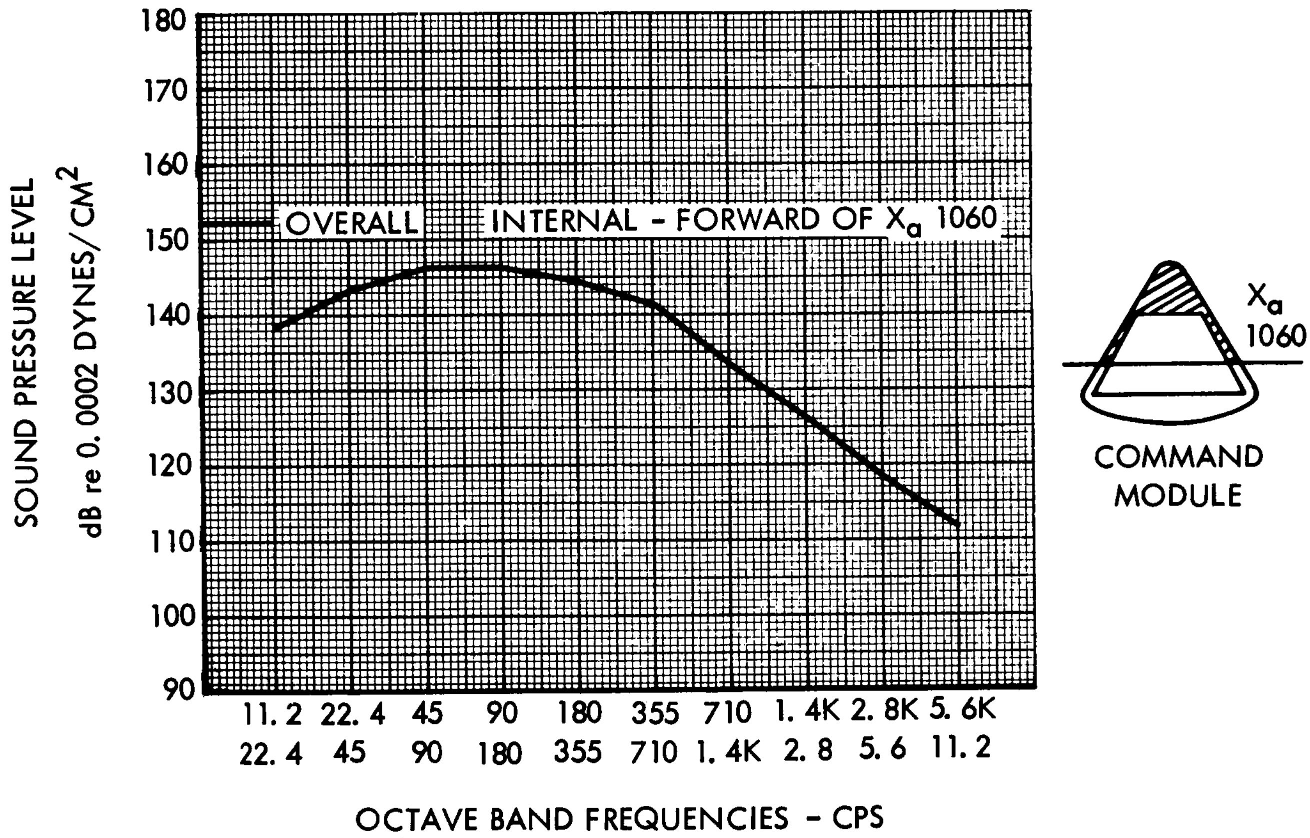

Figure 24. Acoustics CM - Atmospheric Flight - Internal - Forward Xa1060

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ACOUSTICS

COMMAND MODULE
ATMOSPHERIC FLIGHT

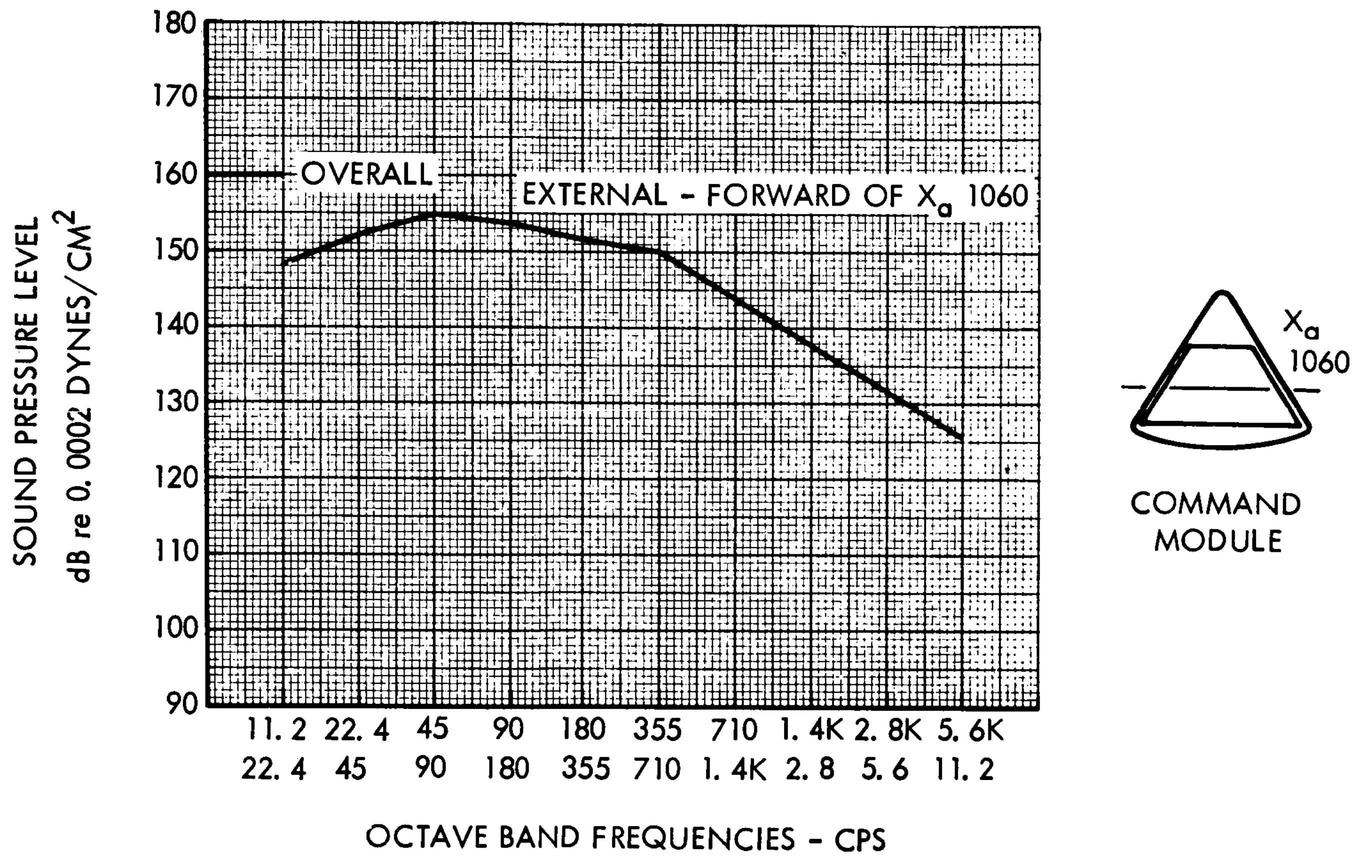

Figure 25. Acoustics CM - Atmospheric Flight - External - Forward Xa1060

ACOUSTICS

COMMAND MODULE

ATMOSPHERIC FLIGHT

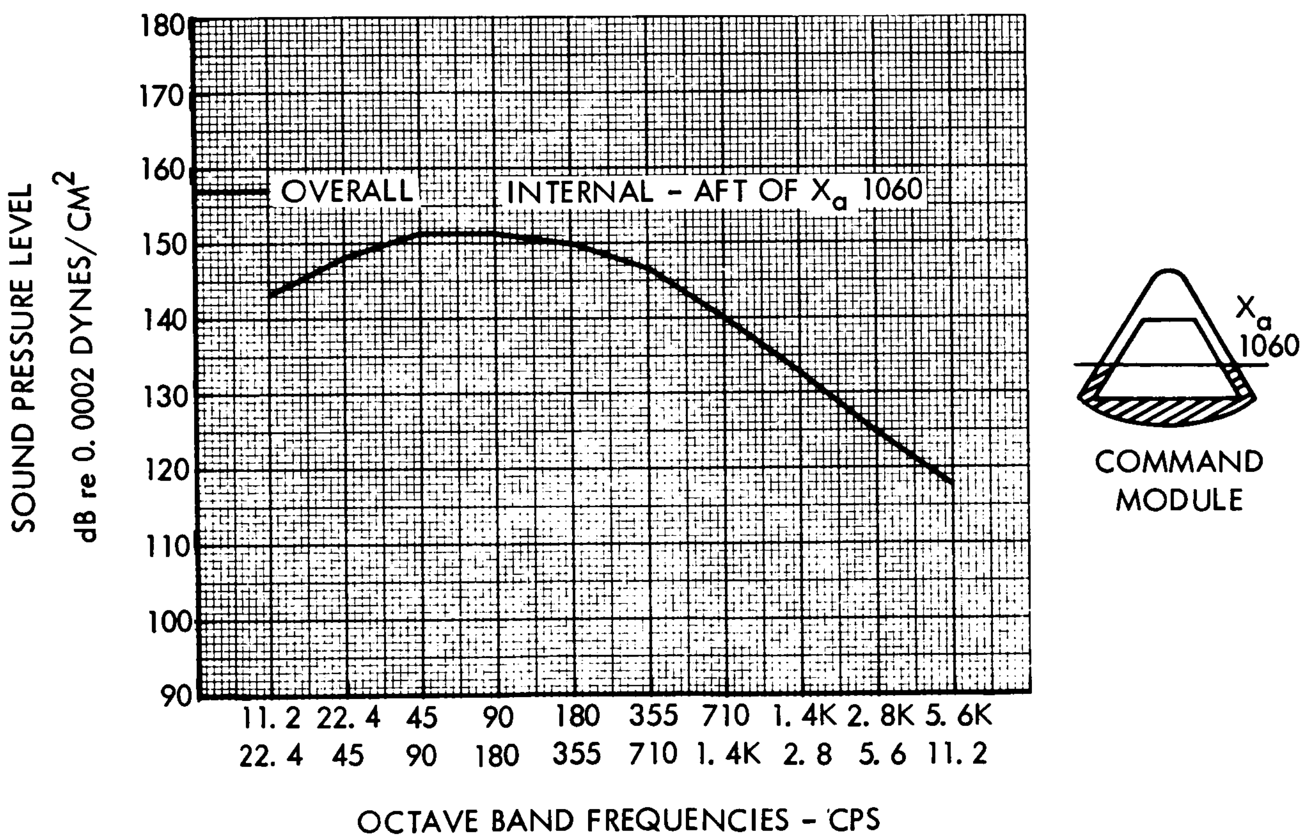

Figure 26. Acoustics CM - Atmospheric Flight - Internal - Aft Xa1060

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ACOUSTICS

COMMAND MODULE
ATMOSPHERIC FLIGHT

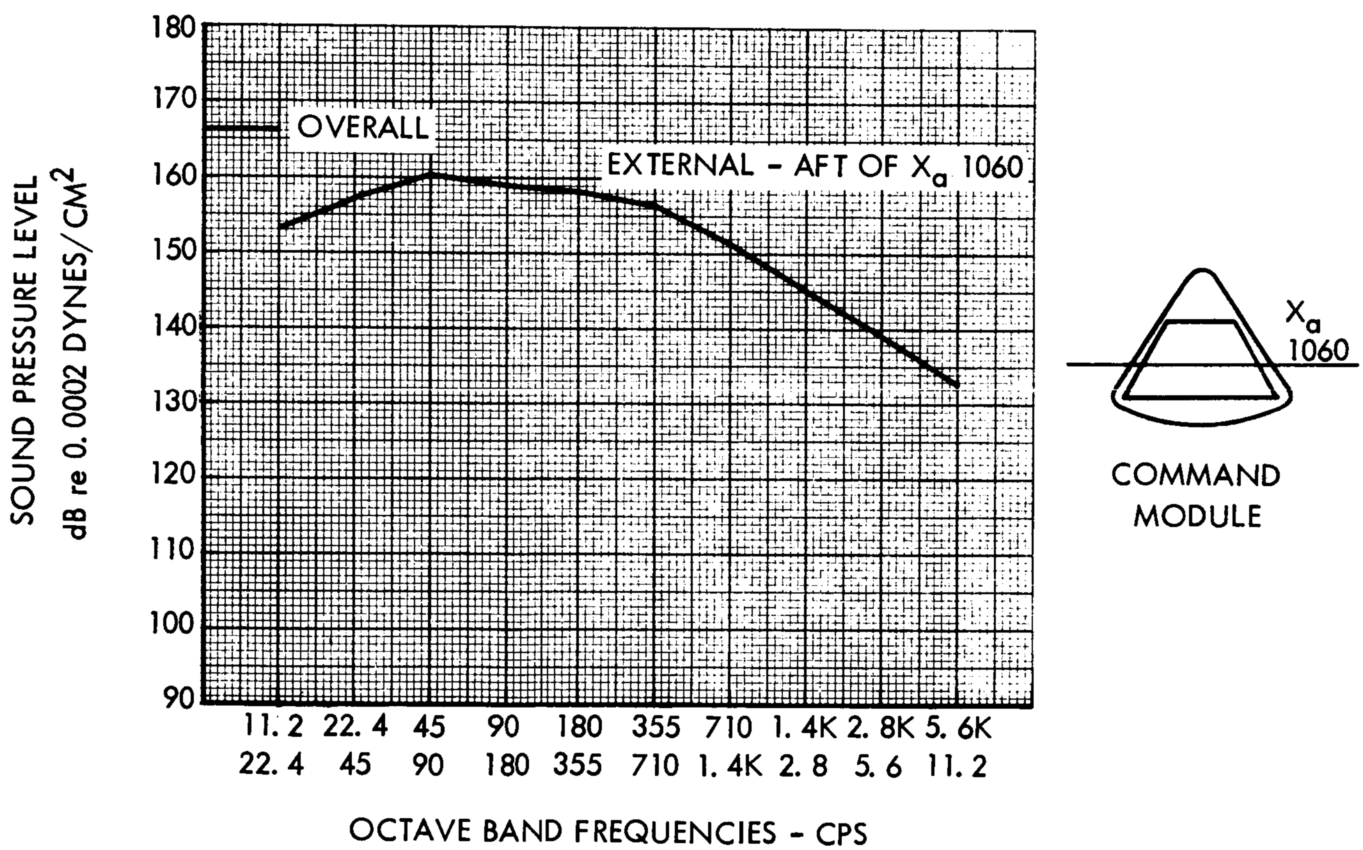

Figure 27. Acoustics CM - Atmospheric Flight - External - Aft Xa1060

ACOUSTICS
COMMAND MODULE
ATMOSPHERIC FLIGHT

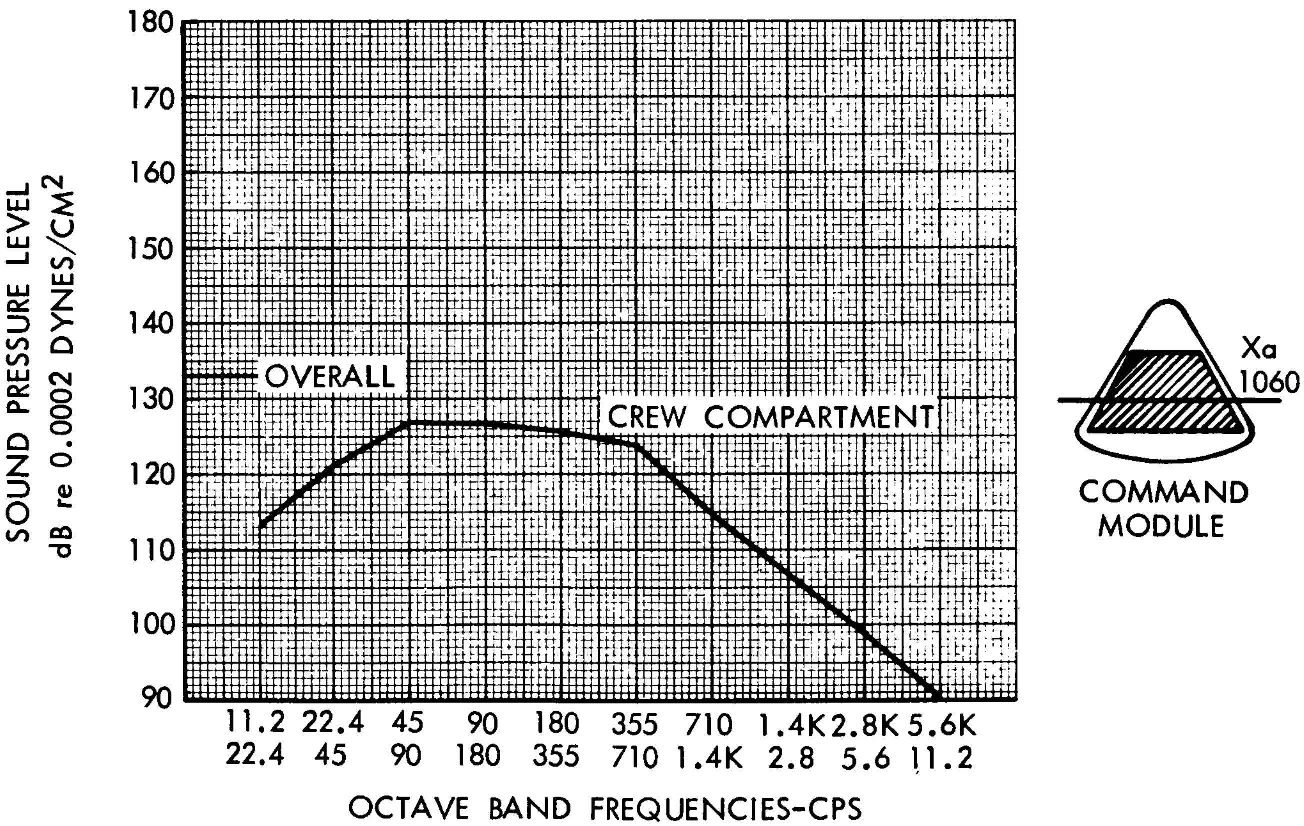

Figure 28. Acoustics CM - Atmospheric Flight - Crew Compartment

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ACOUSTICS

SERVICE MODULE
ATMOSPHERIC FLIGHT

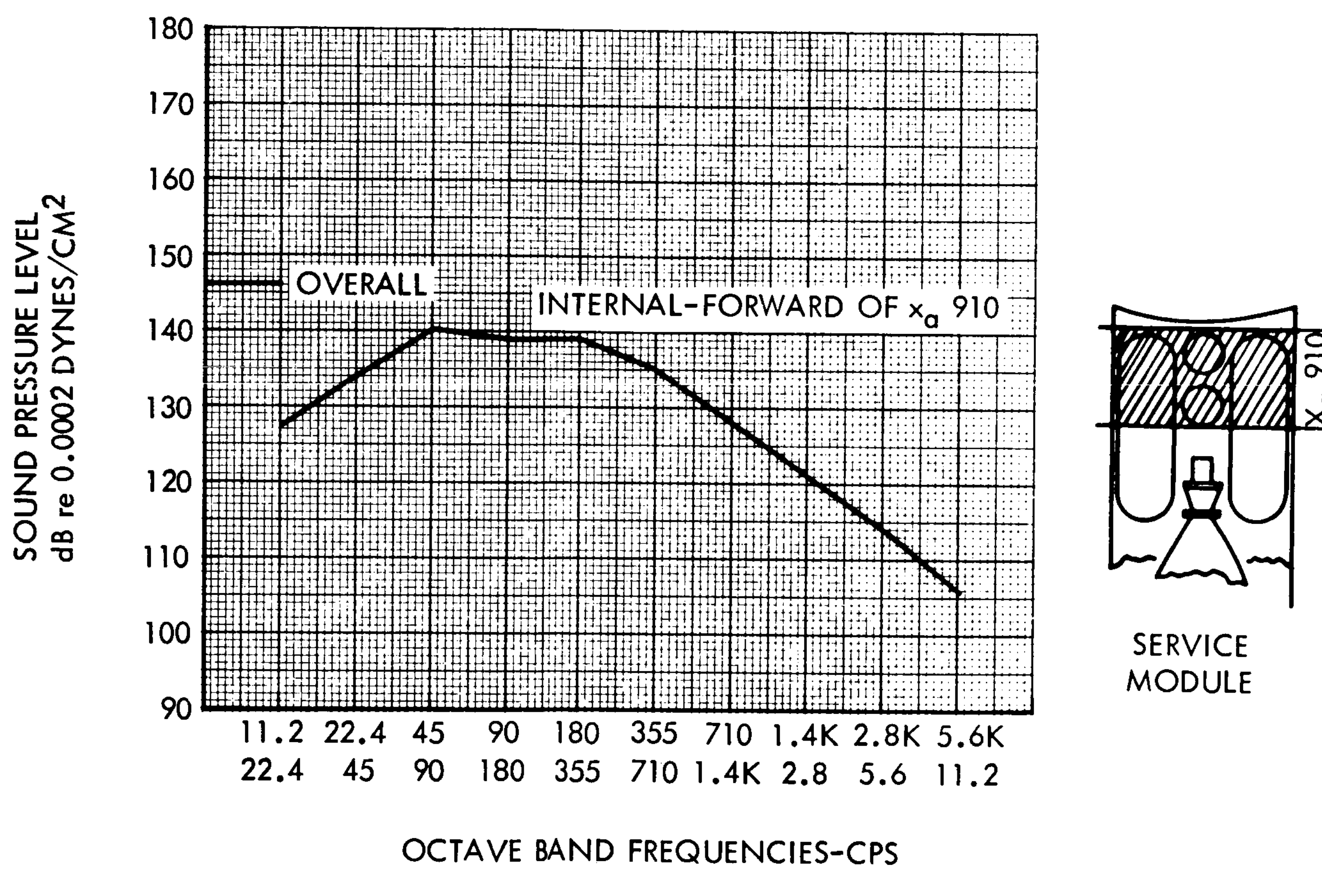
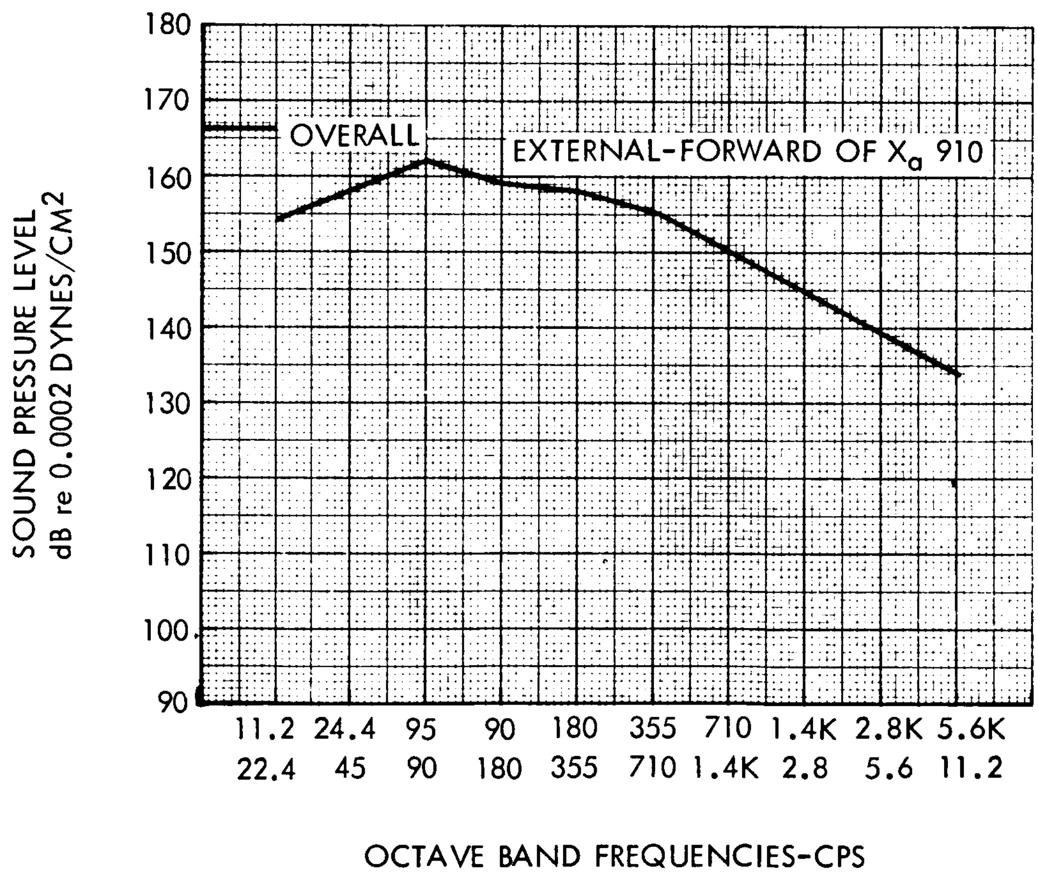


Figure 29. Acoustics SM - Atmospheric Flight - Internal - Forward Xa910

ACOUSTICS

SERVICE MODULE
ATMOSPHERIC FLIGHT



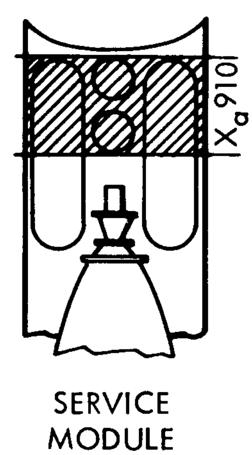


Figure 30. Acoustics SM - Atmospheric Flight - External - Forward Xa910

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ACOUSTICS

SM RCS ENGINE & PANEL
ATMOSPHERIC FLIGHT

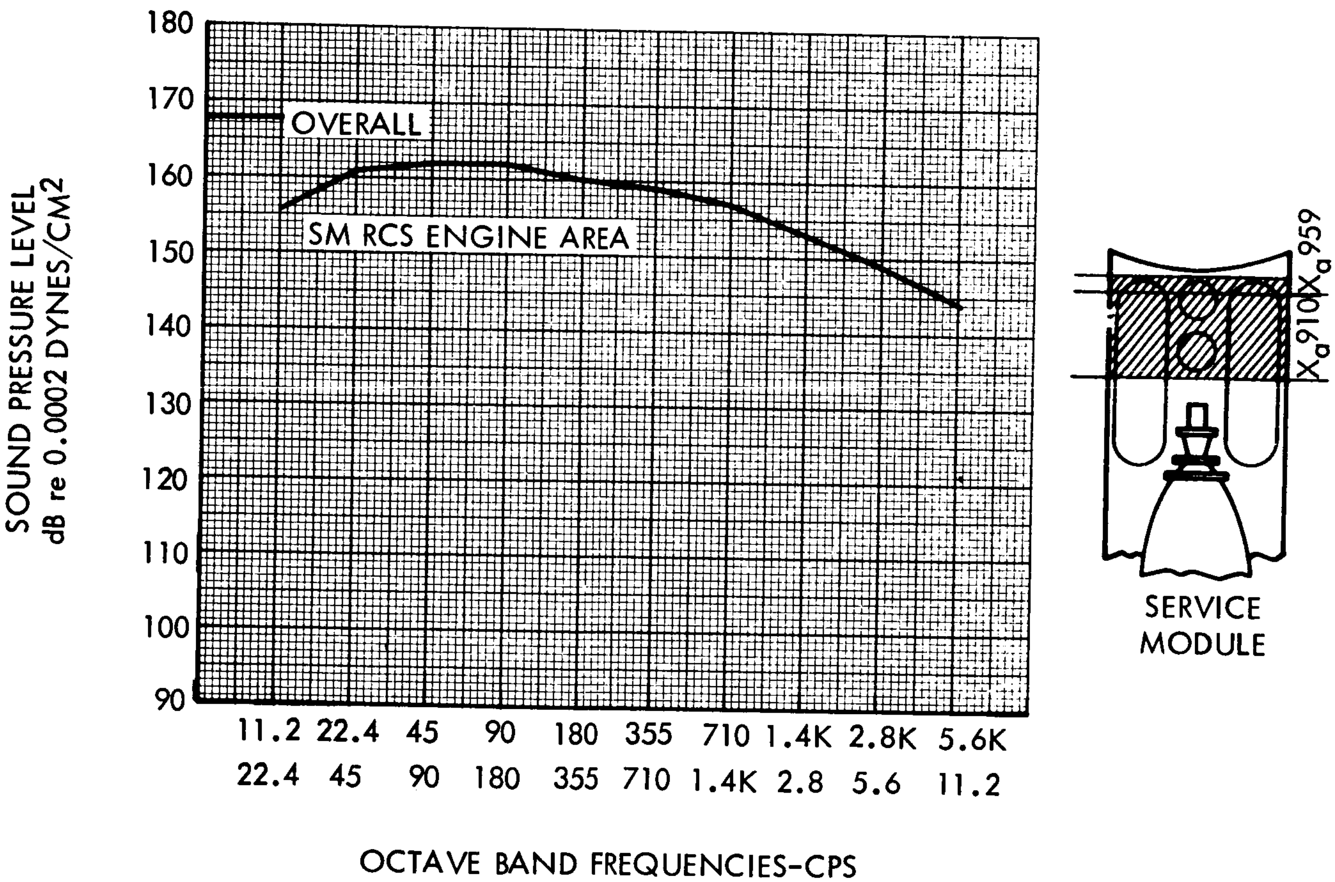


Figure 31. Acoustics SM/RCS Engine & Panel - Atmospheric Flight

~~CONFIDENTIAL~~

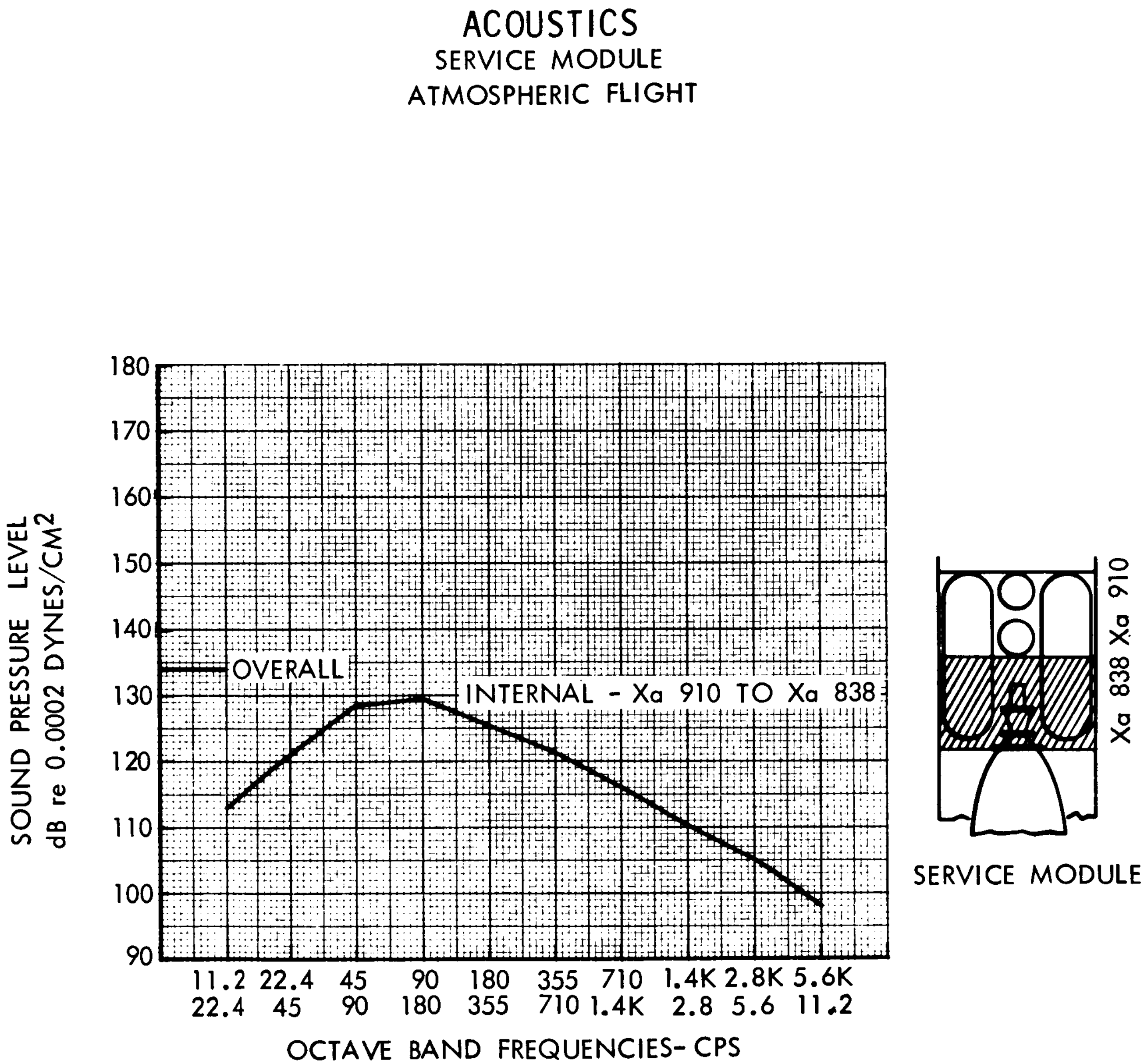


Figure 32. Acoustics SM - Atmospheric Flight - Internal XA910 to Xa838

~~CONFIDENTIAL~~

ACOUSTICS
SERVICE MODULE
ATMOSPHERIC FLIGHT

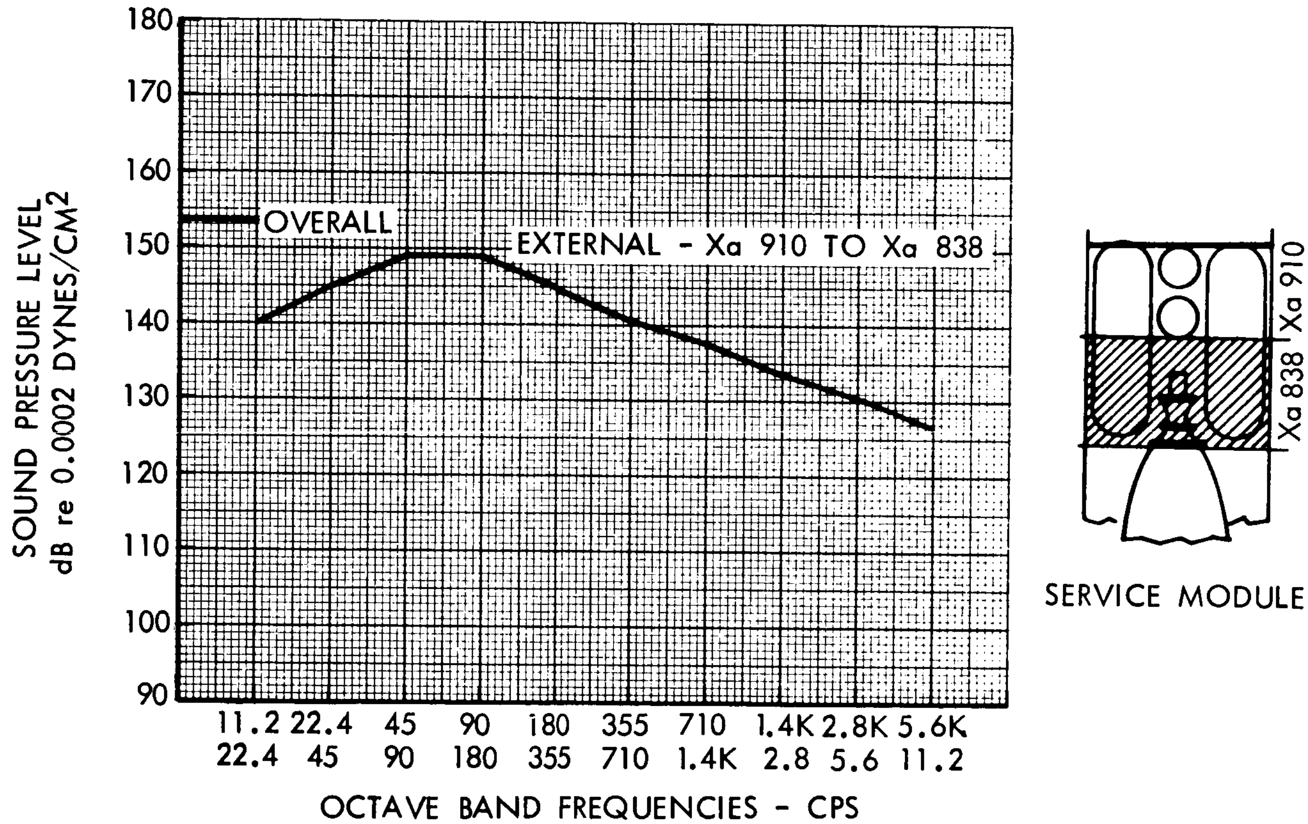
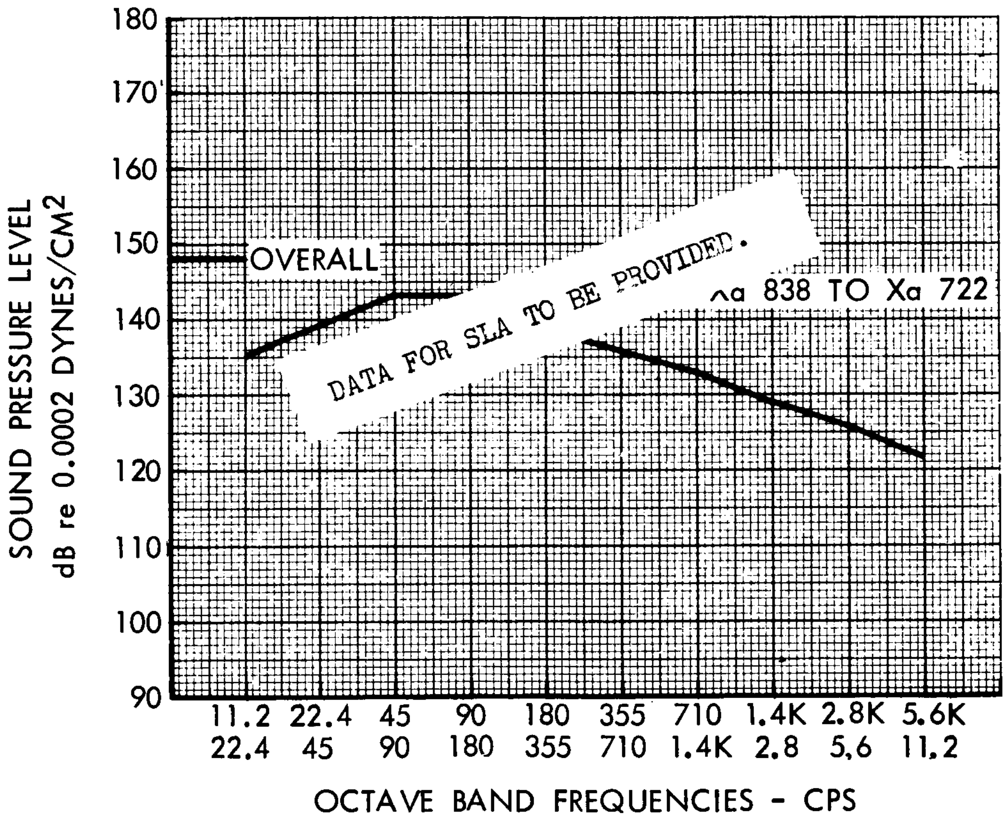


Figure 33. Acoustics SM - Atmospheric Flight - External Xa910 to Xa838

~~CONFIDENTIAL~~

ACOUSTICS
ADAPTER
ATMOSPHERIC FLIGHT



ADAPTER

Figure 34. Acoustics Adapter - Atmospheric Flight - External Xa838 to Xa722

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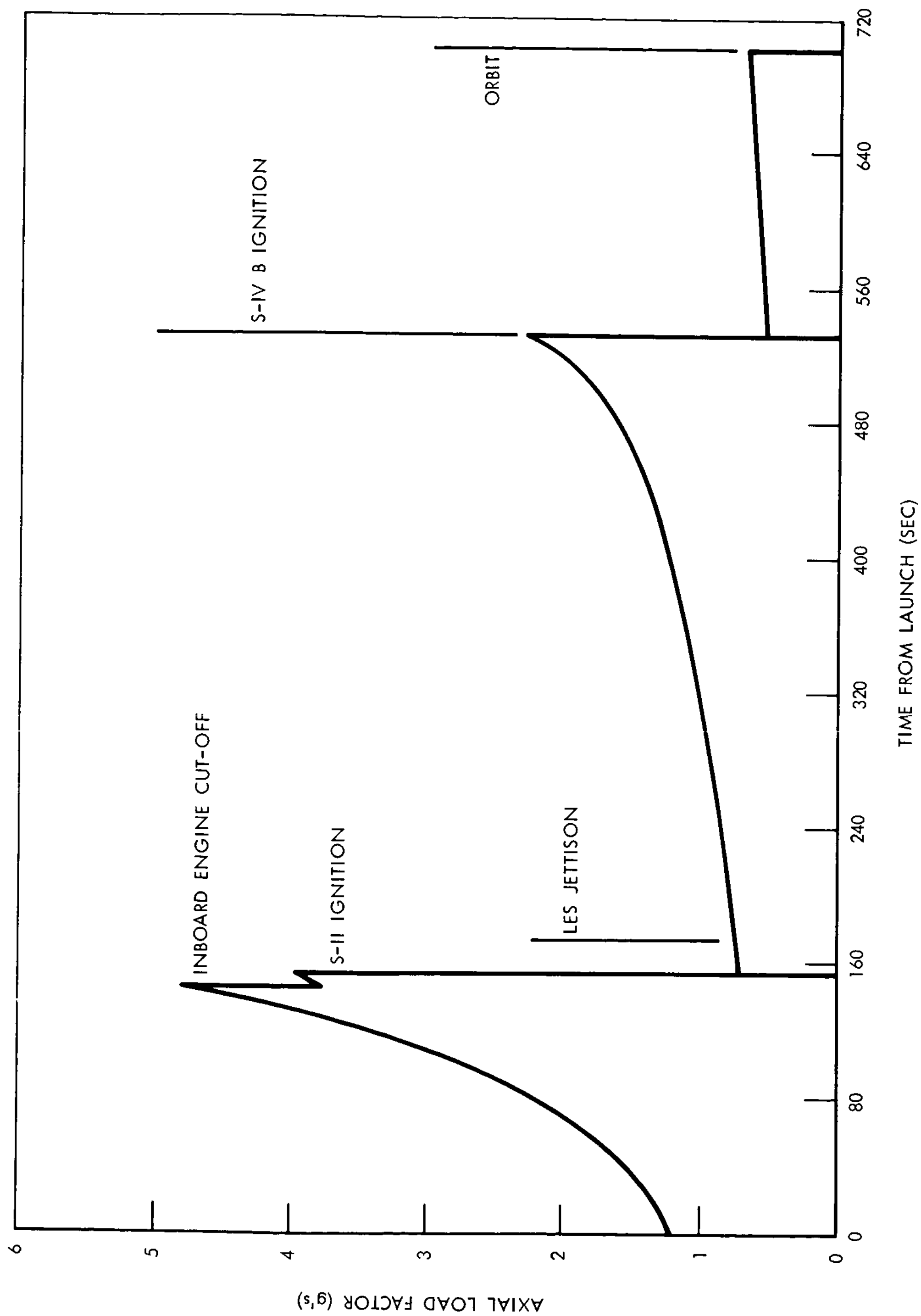

Figure 35. Axial Acceleration - Nominal Saturn V Boost

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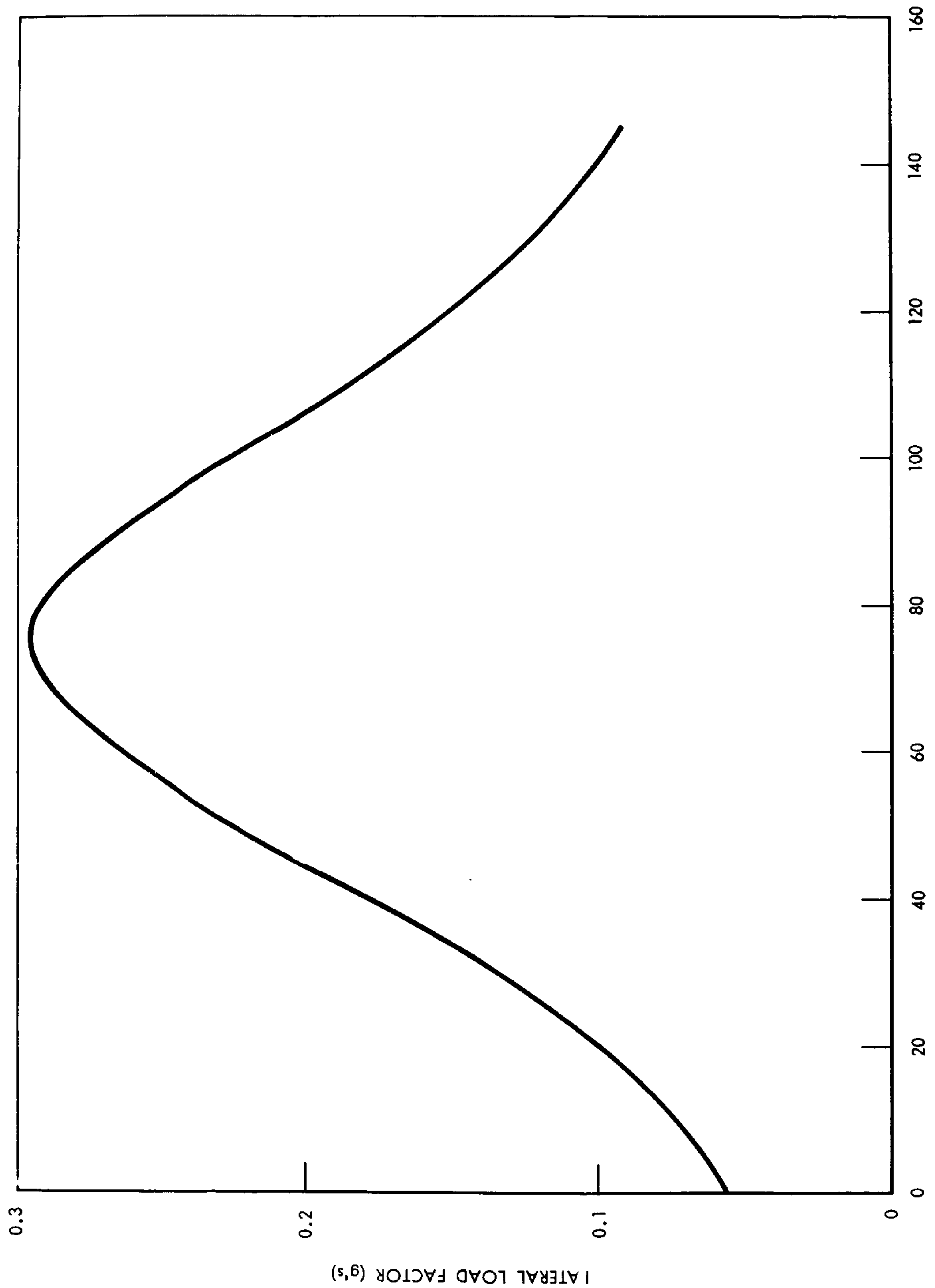

Figure 36. Lateral Acceleration During First Stage Boost

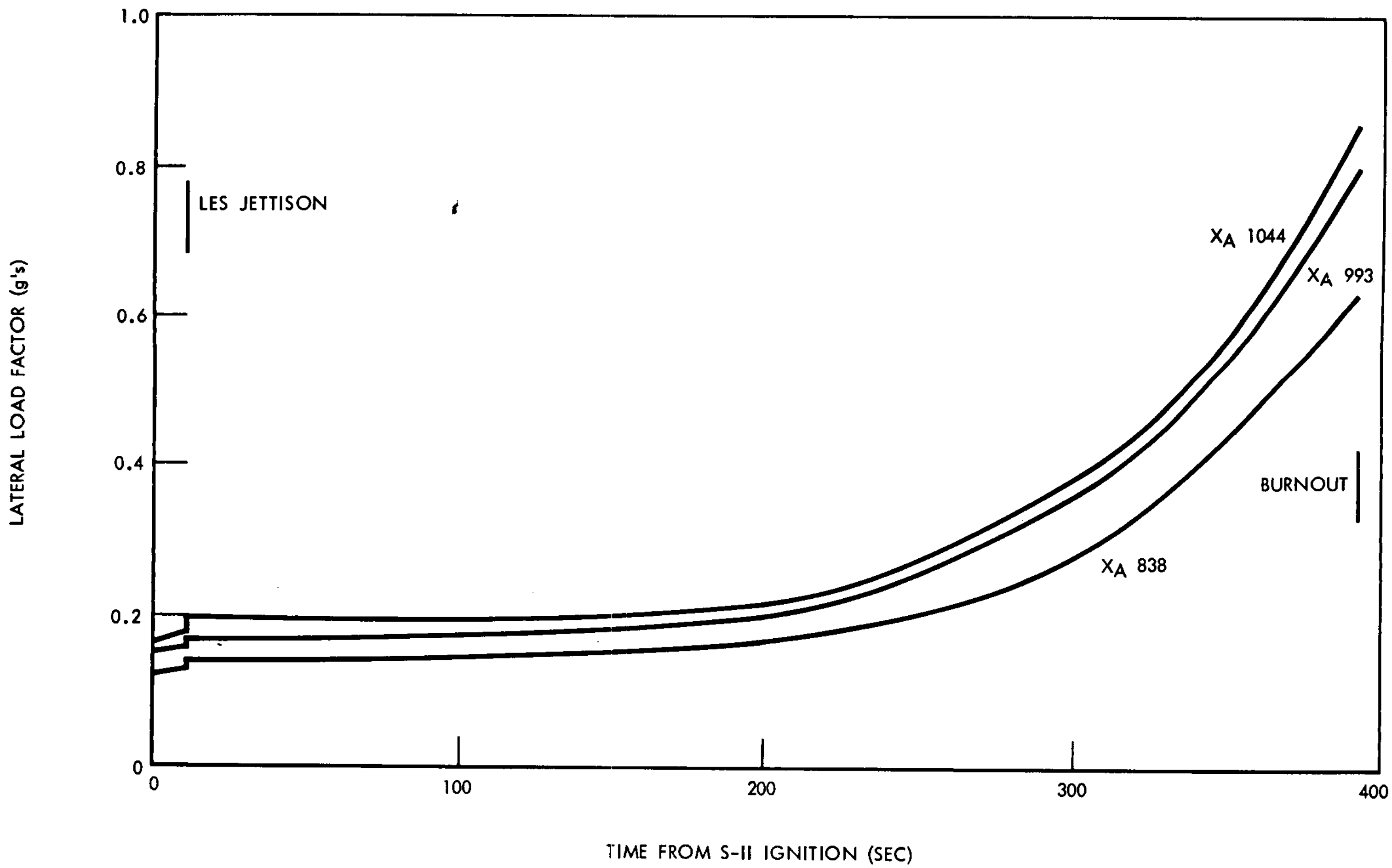

Figure 37. Lateral Acceleration - S-II Flight at Maximum Gimbal Deflection

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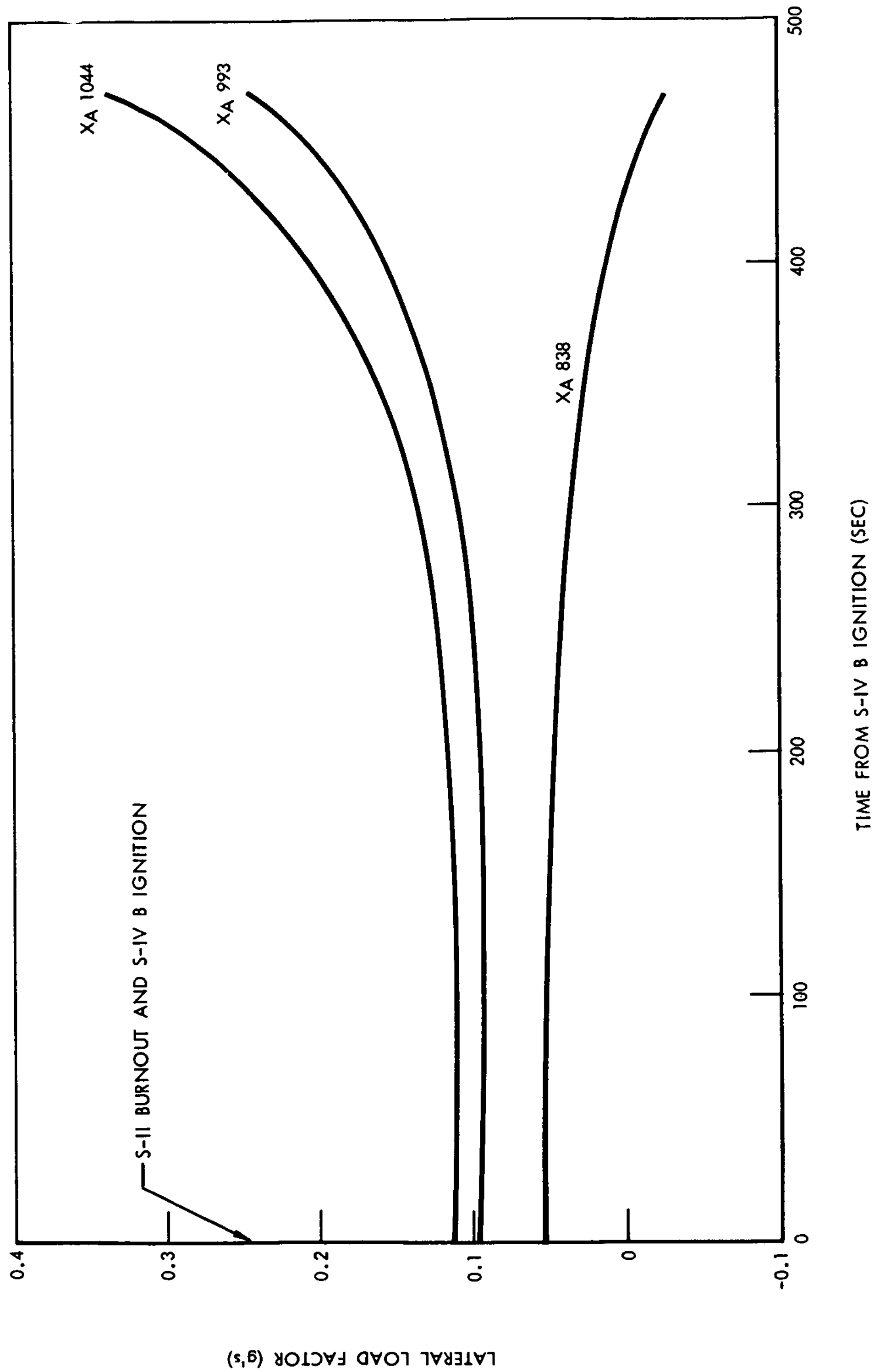

Figure 38. Lateral Acceleration - S-IVB Flight at Maximum Gimbal Deflection

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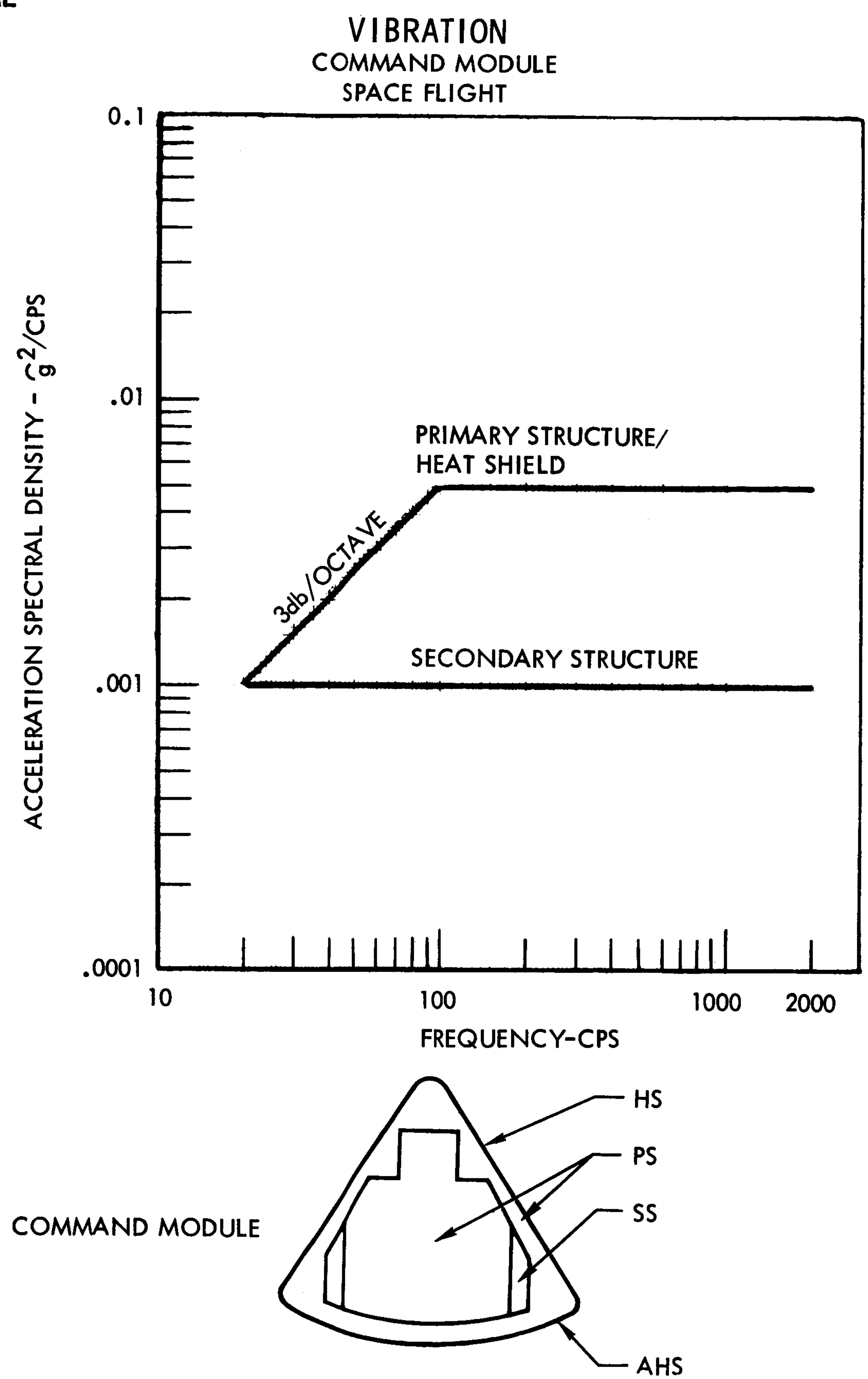

Figure 39. Vibration CM - Space Flight

VIBRATION
SERVICE MODULE
SPACE FLIGHT

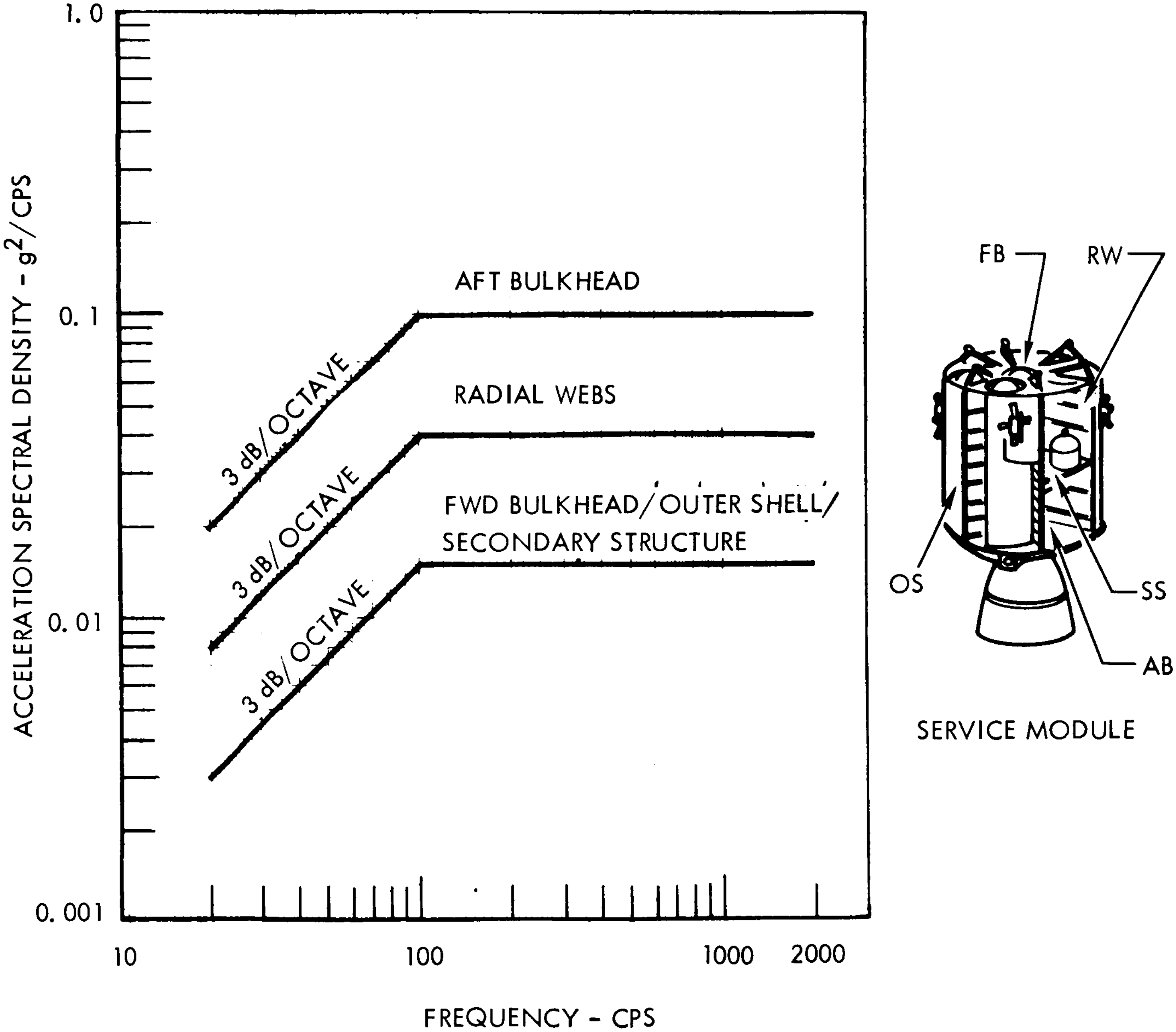

Figure 40. Vibration SM - Space Flight

CO[redacted]

NOTE: Referenced axis is at the equipment and also parallel to spacecraft's major axis

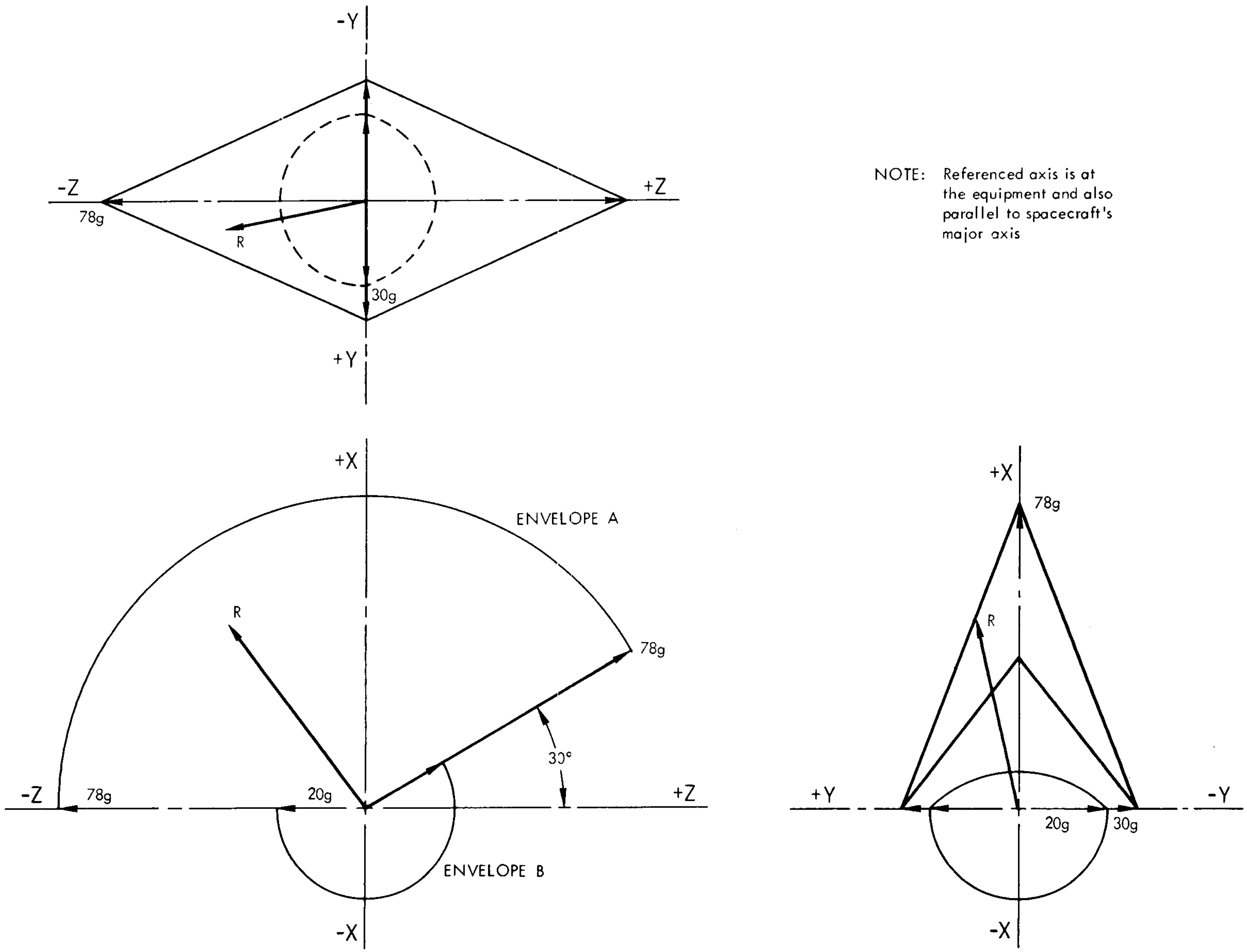

Figure 41. Internal Equipment Ultimate Design **Accelerations Diagram I**

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[redacted]

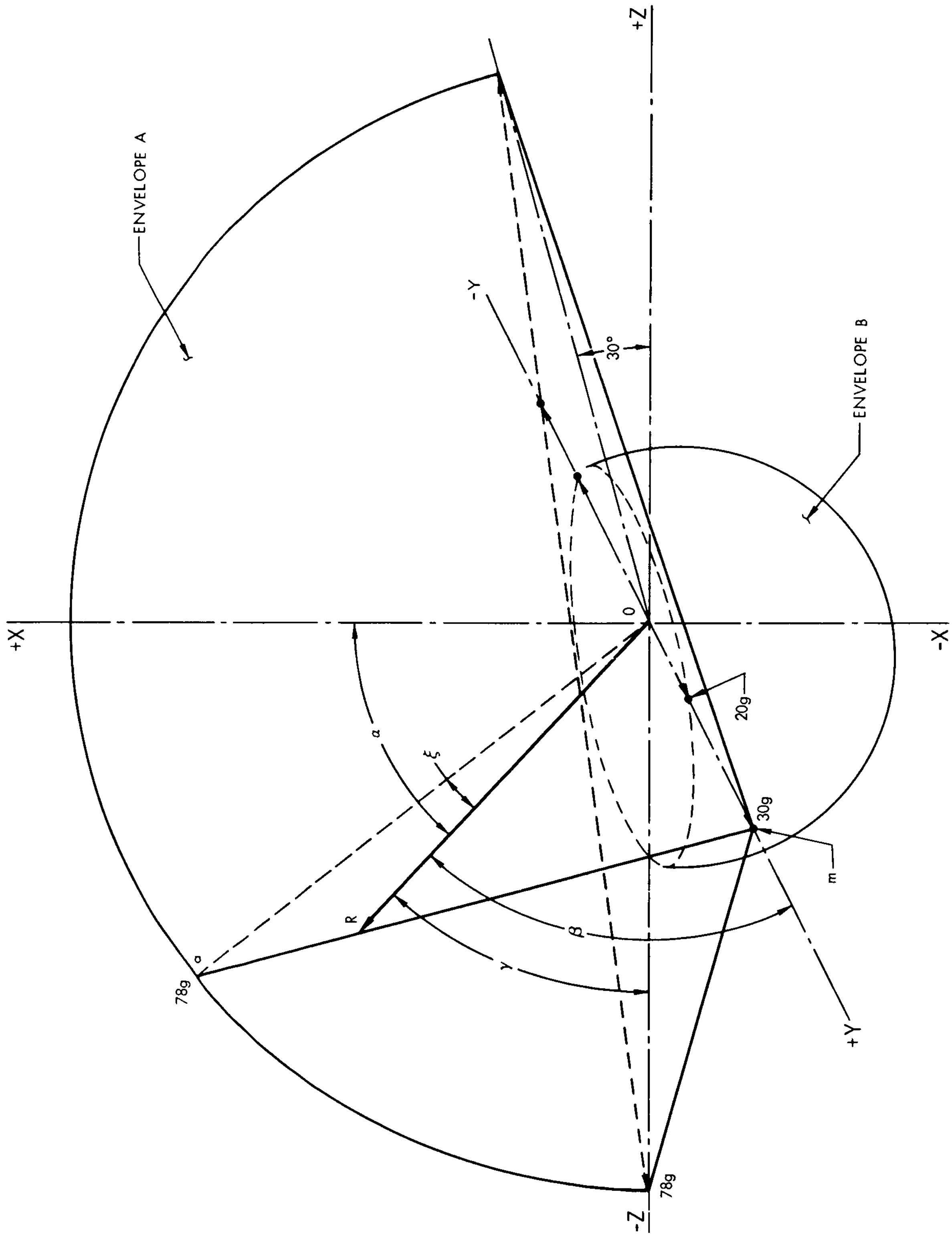

Figure 42. Internal Equipment Ultimate Design Accelerations Diagram II

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VIBRATION
LAUNCH ESCAPE SYSTEM

HIGH "Q" ABORT

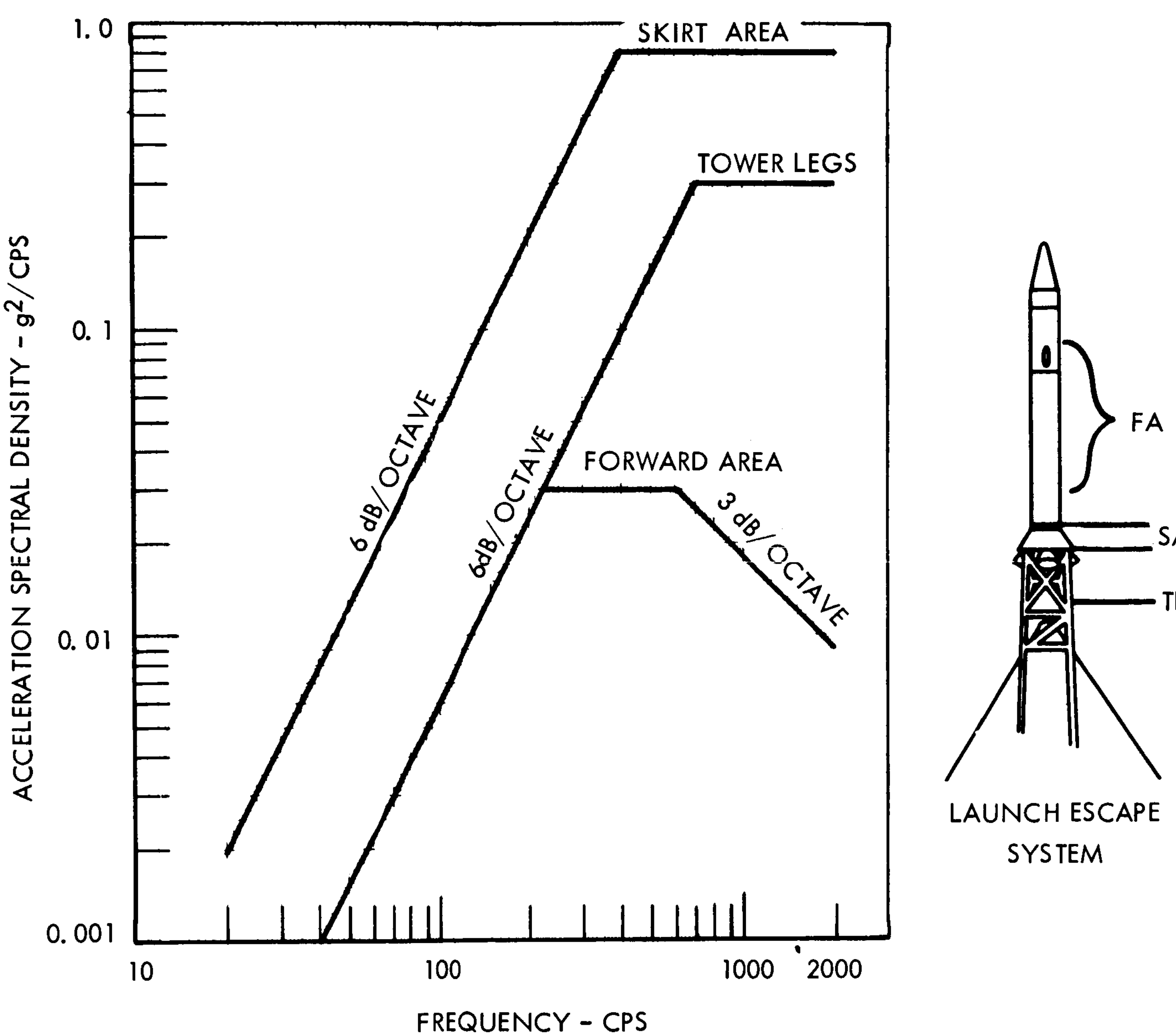

Figure 43. Vibration LES - High "Q" Abort

~~CONFIDENTIAL~~

VIBRATION
COMMAND MODULE
HIGH "Q" ABORT

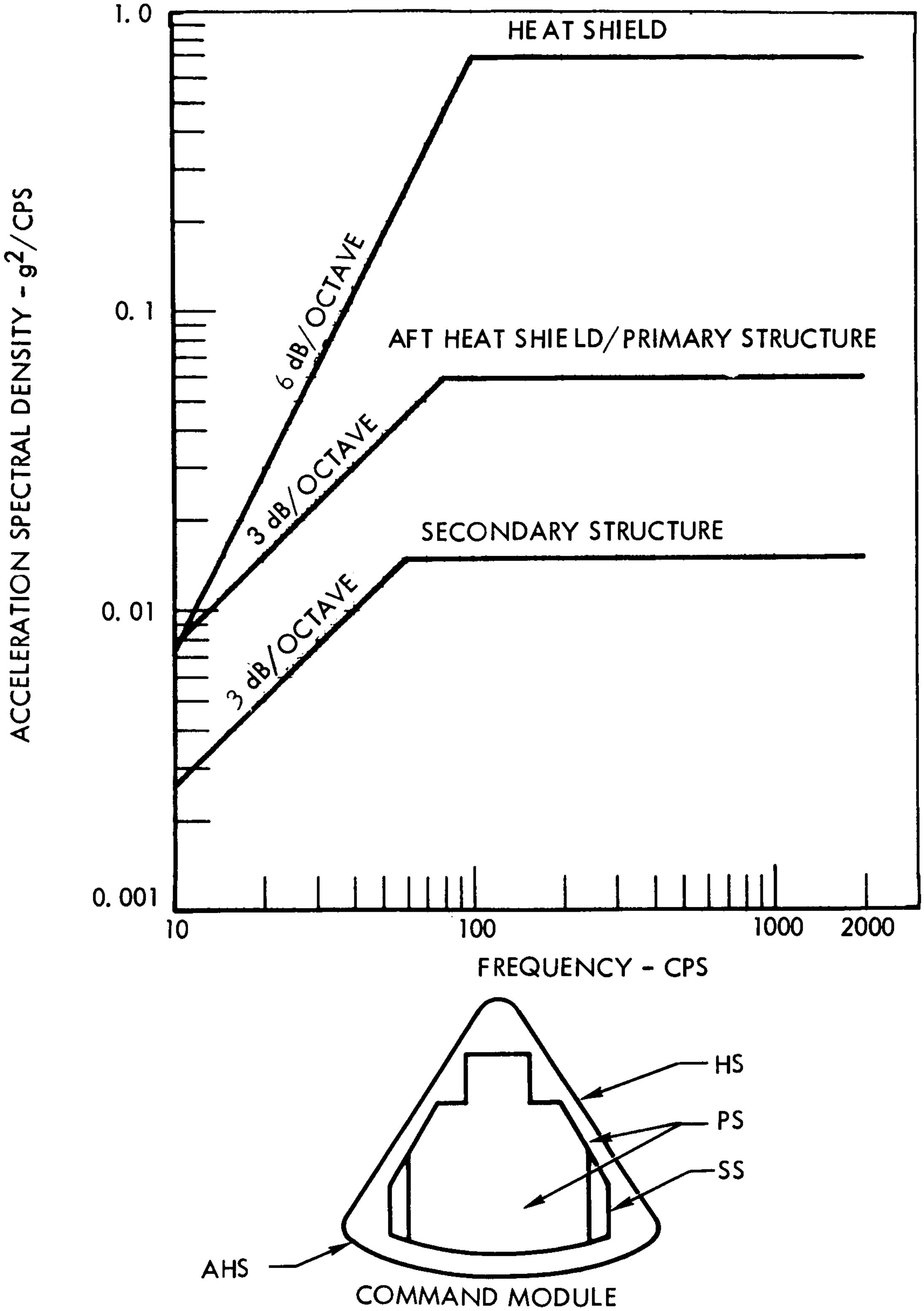

Figure 44. Vibration CM - High "Q" Abort

~~CONFIDENTIAL~~

ACOUSTICS
LAUNCH ESCAPE SYSTEM
HIGH "Q" ABORT

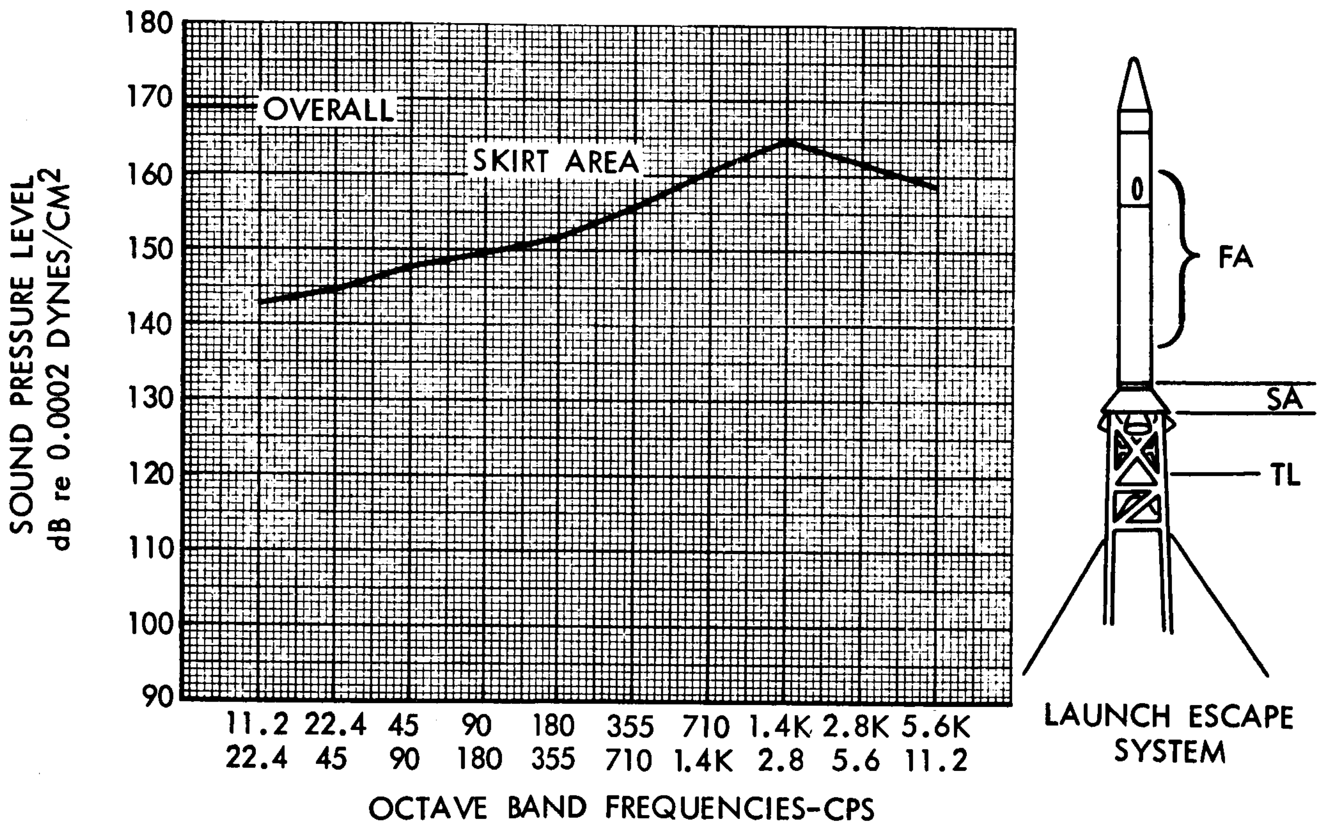

Figure 45. Acoustics LES - High "Q" Abort

ACOUSTICS
COMMAND MODULE
HIGH "Q" ABORT

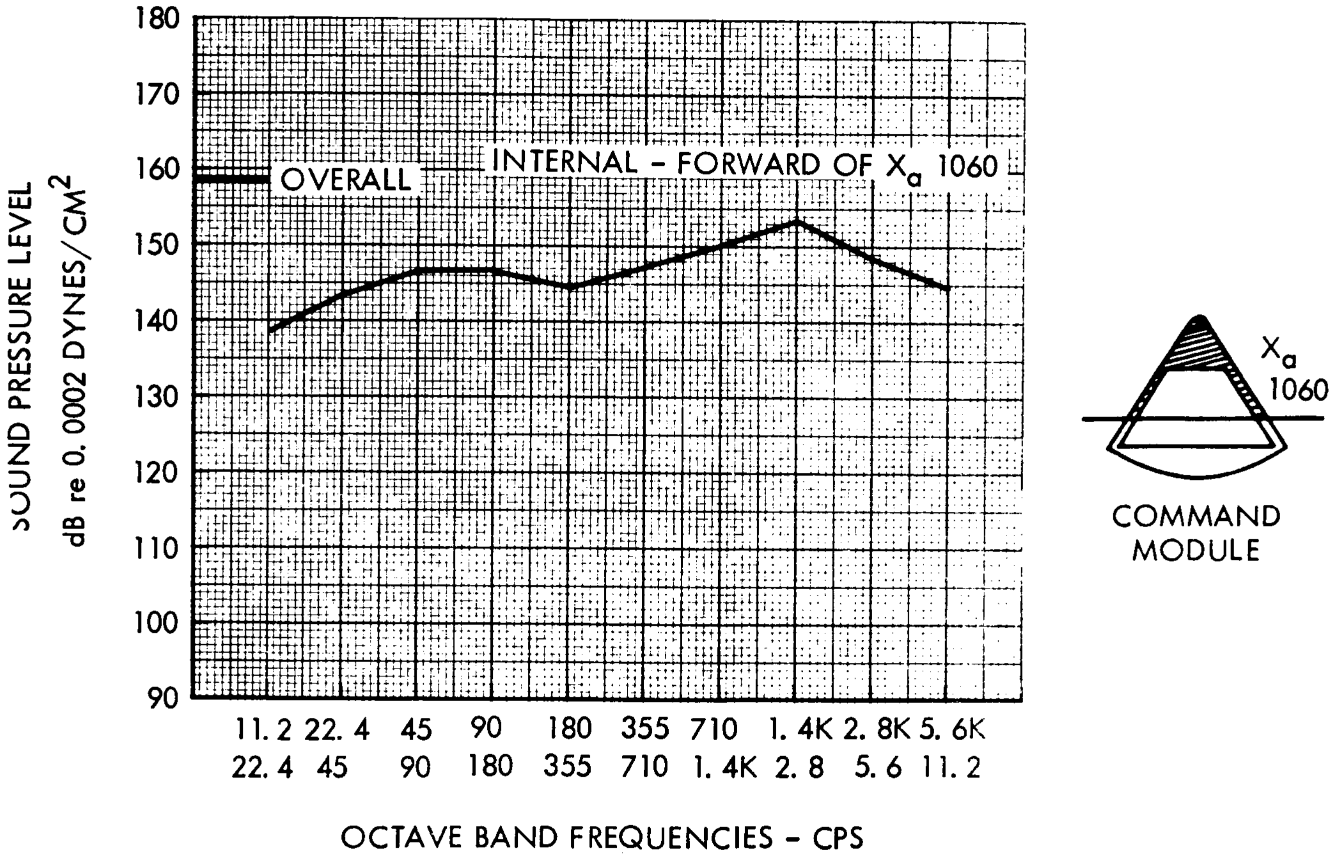

Figure 46. Acoustics CM - High "Q" Abort - Internal - Forward Xa1060

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ACOUSTICS

COMMAND MODULE
HIGH "Q" ABORT

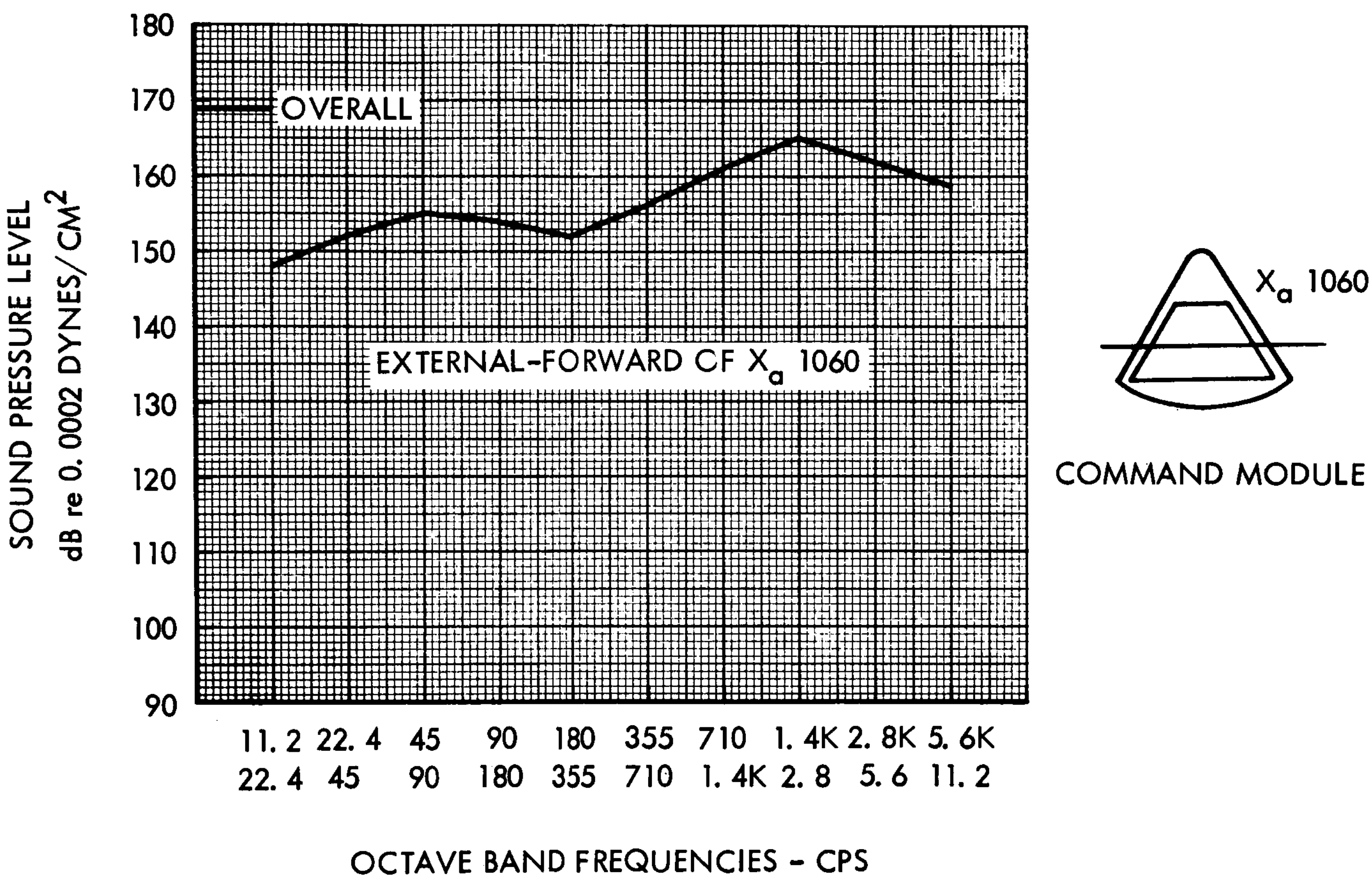
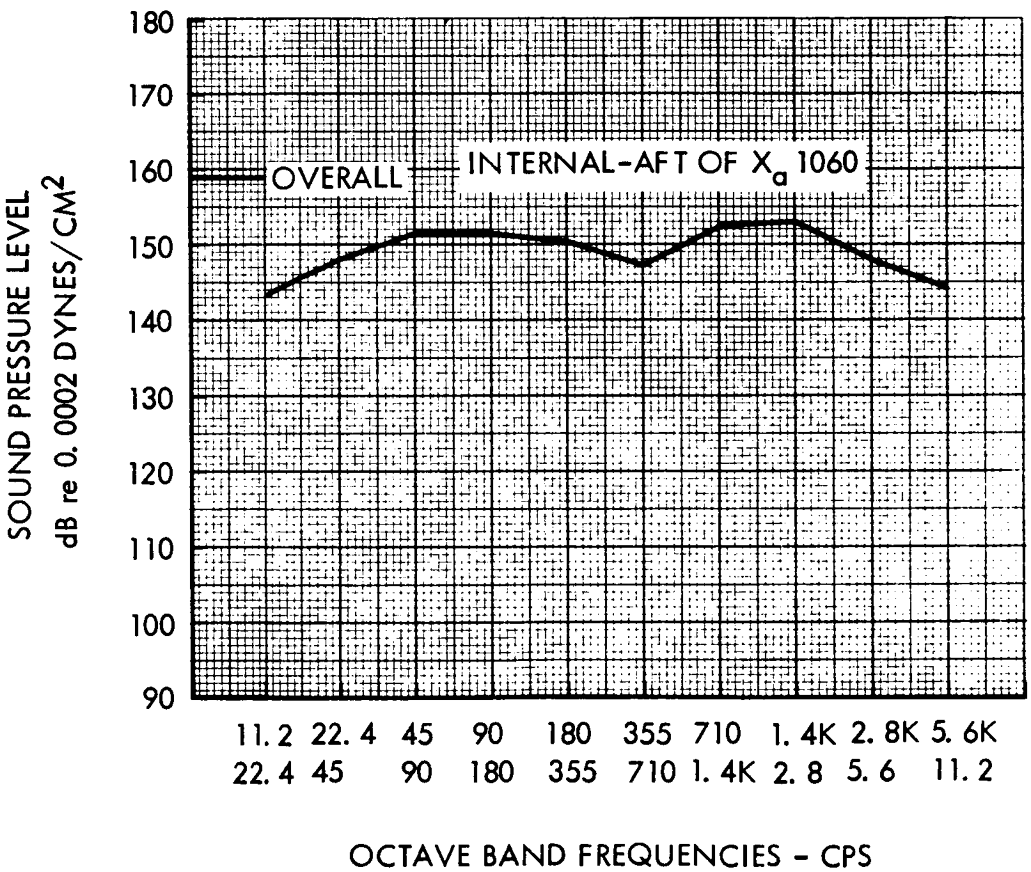
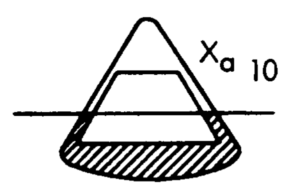


Figure 47. Acoustics CM - High "Q" Abort - External - Forward of X1060

ACOUSTICS

COMMAND MODULE
HIGH "Q" ABORT





COMMAND
MODULE

Figure 48. Acoustics CM - High "Q" Abort - Internal - Aft of Xa1060

ACOUSTICS
COMMAND MODULE
HIGH "Q" ABORT

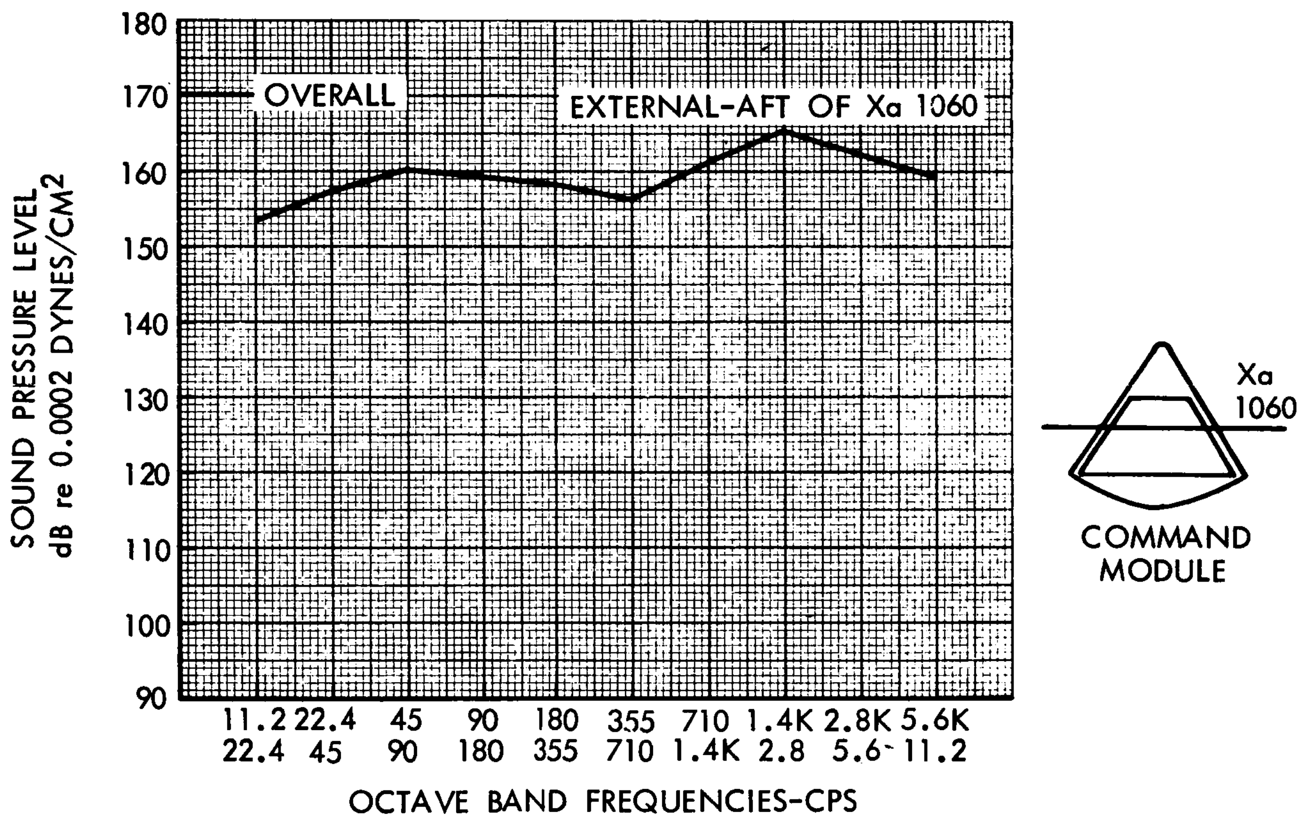
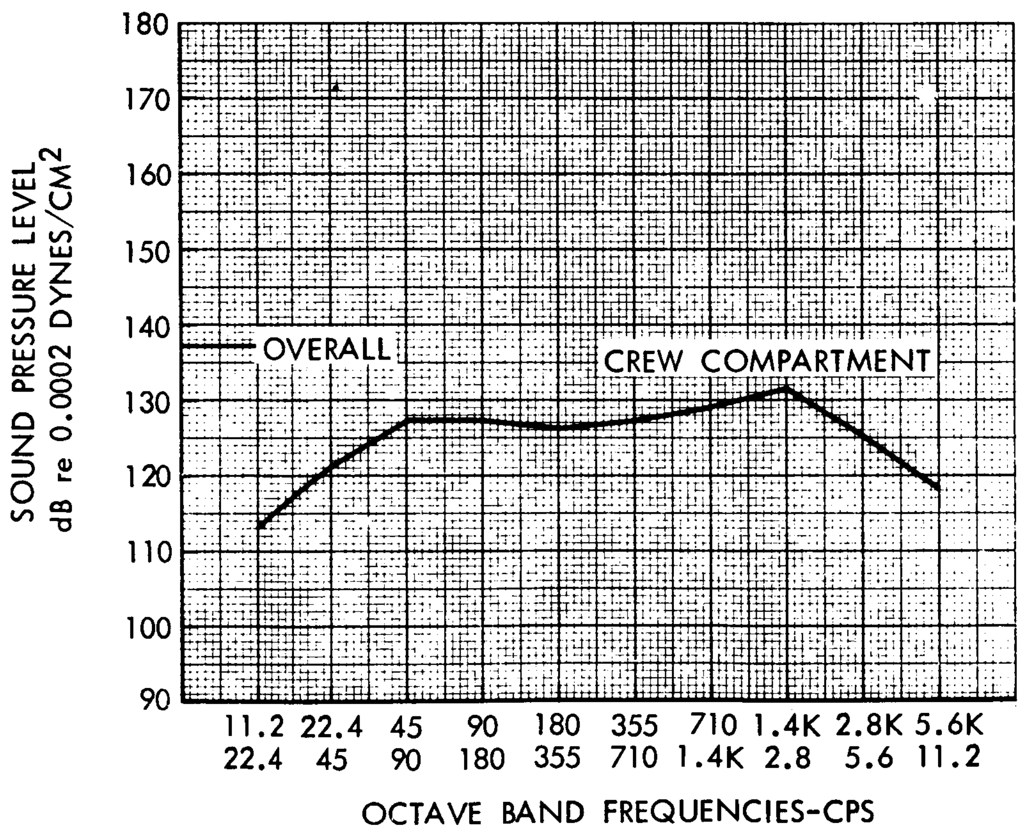


Figure 49. Acoustics CM - High "Q" Abort - External - Aft of Xa1060

ACOUSTICS
COMMAND MODULE
HIGH "Q" ABORT



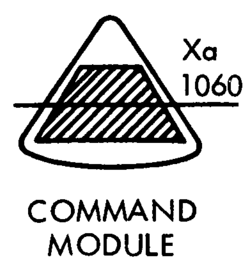


Figure 50. Acoustics CM - High "Q" Abort - Crew Compartment

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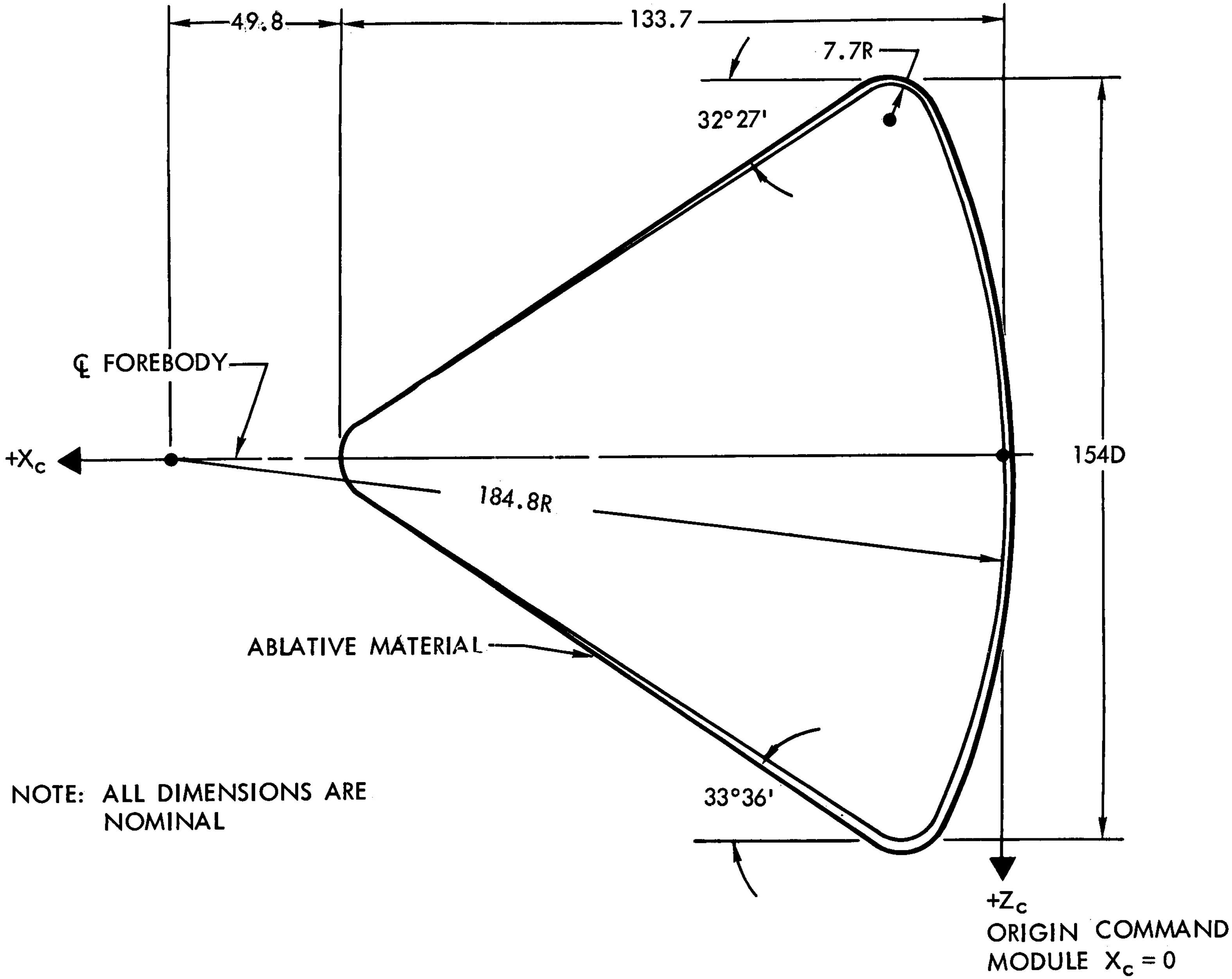

NOTE: ALL DIMENSIONS ARE NOMINAL

Figure 51. Command Module External Dimensions Including Ablative Material

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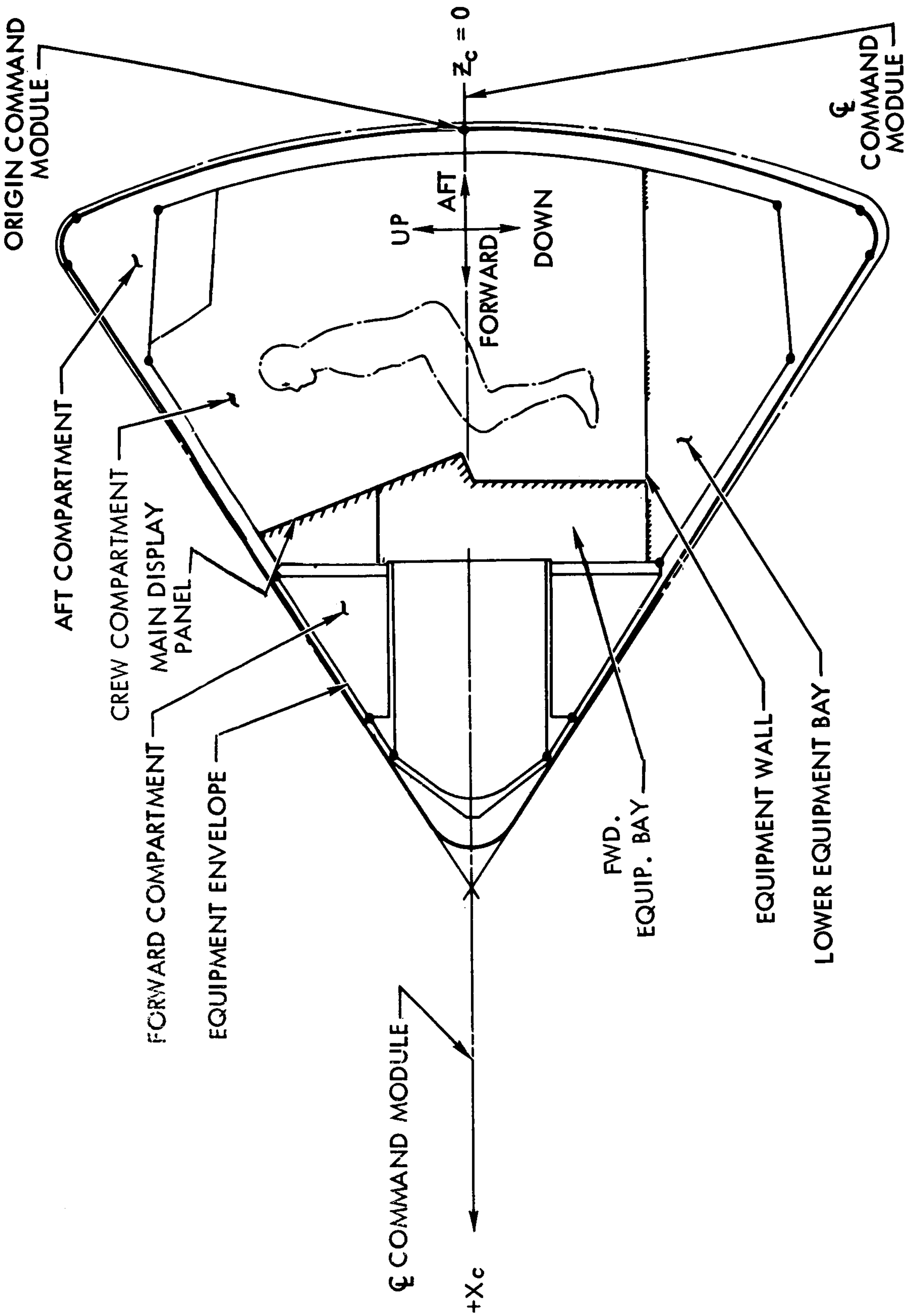

Figure 52. Area Designations - Side View

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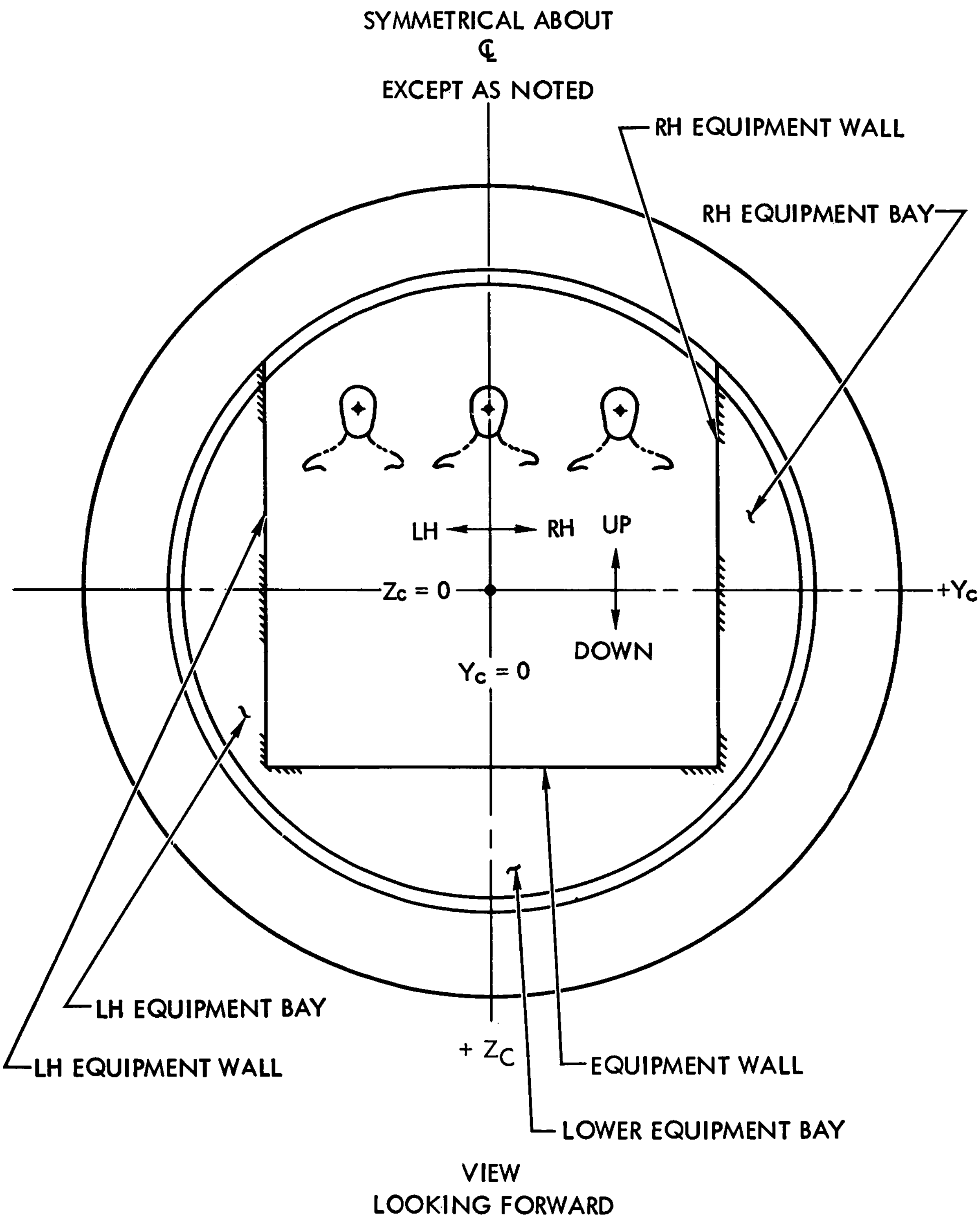
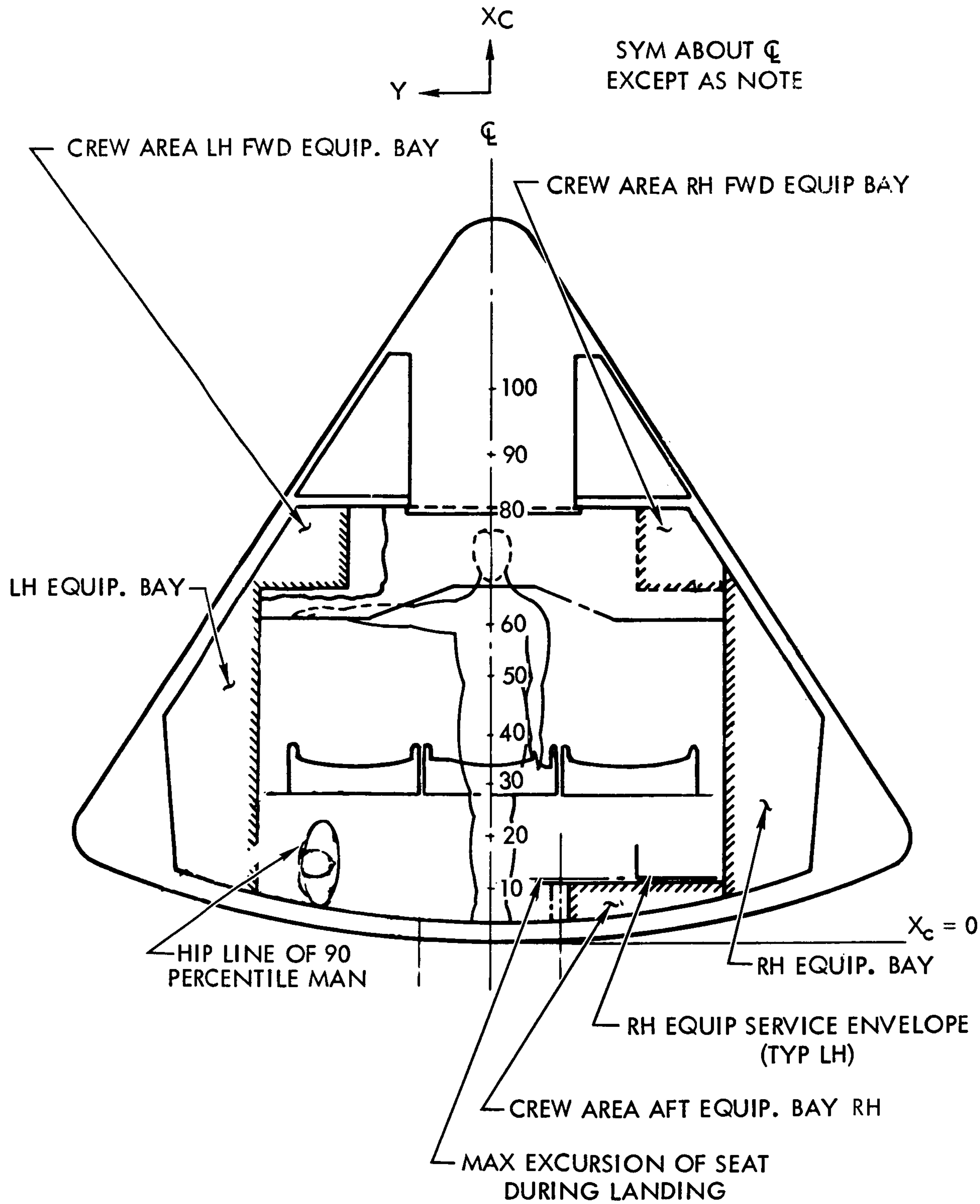


Figure 53. Area Designations - View Looking Forward

~~CONFIDENTIAL~~



VIEW LOOKING TOWARD LOWER END OF CREW AREA

Figure 54. Area Designations - View Looking Toward Lower End of Crew Area

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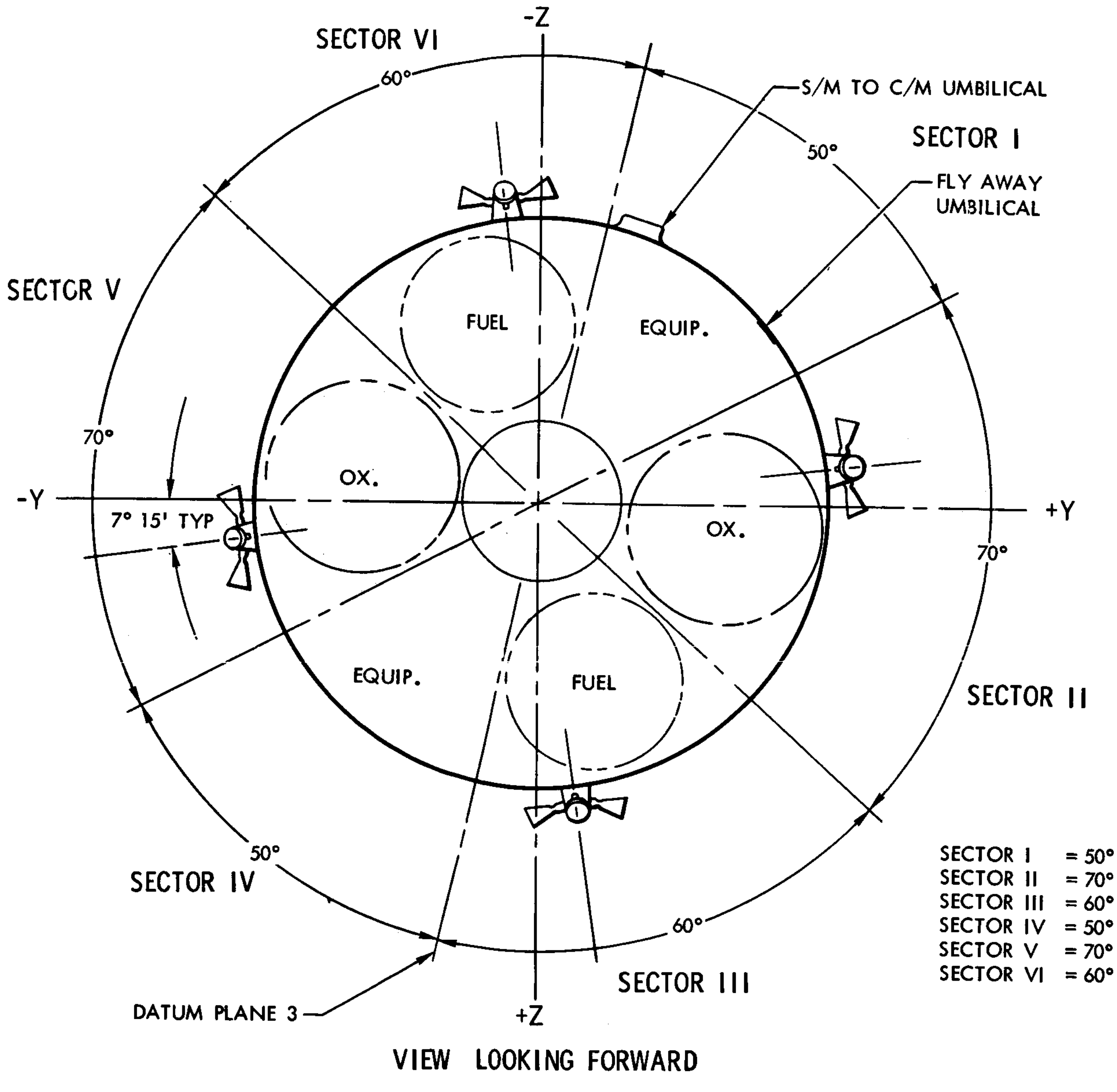

Figure 55. Service Module Inboard Profile - View Looking Forward

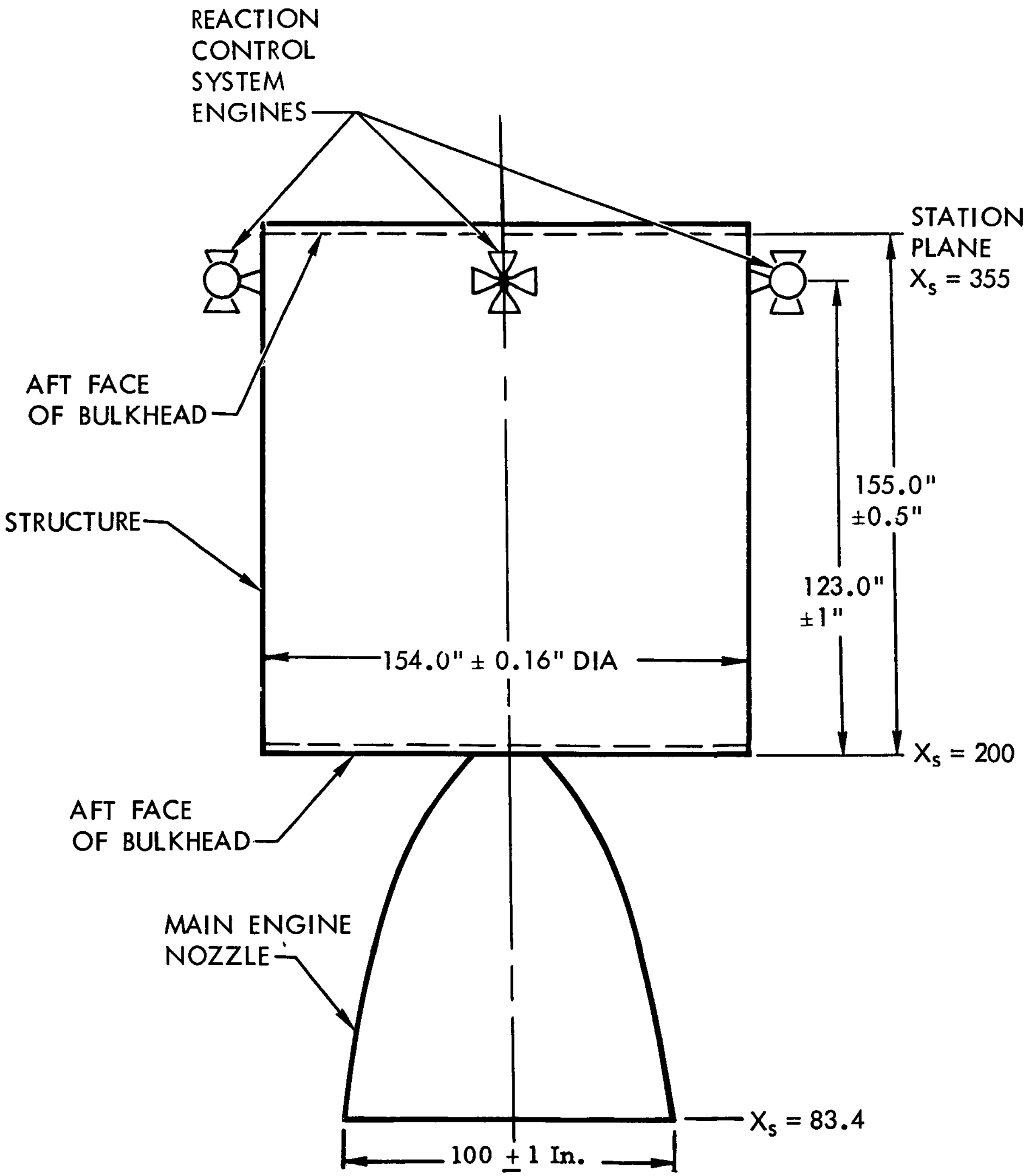

Figure 56. Service Module Inboard Profile

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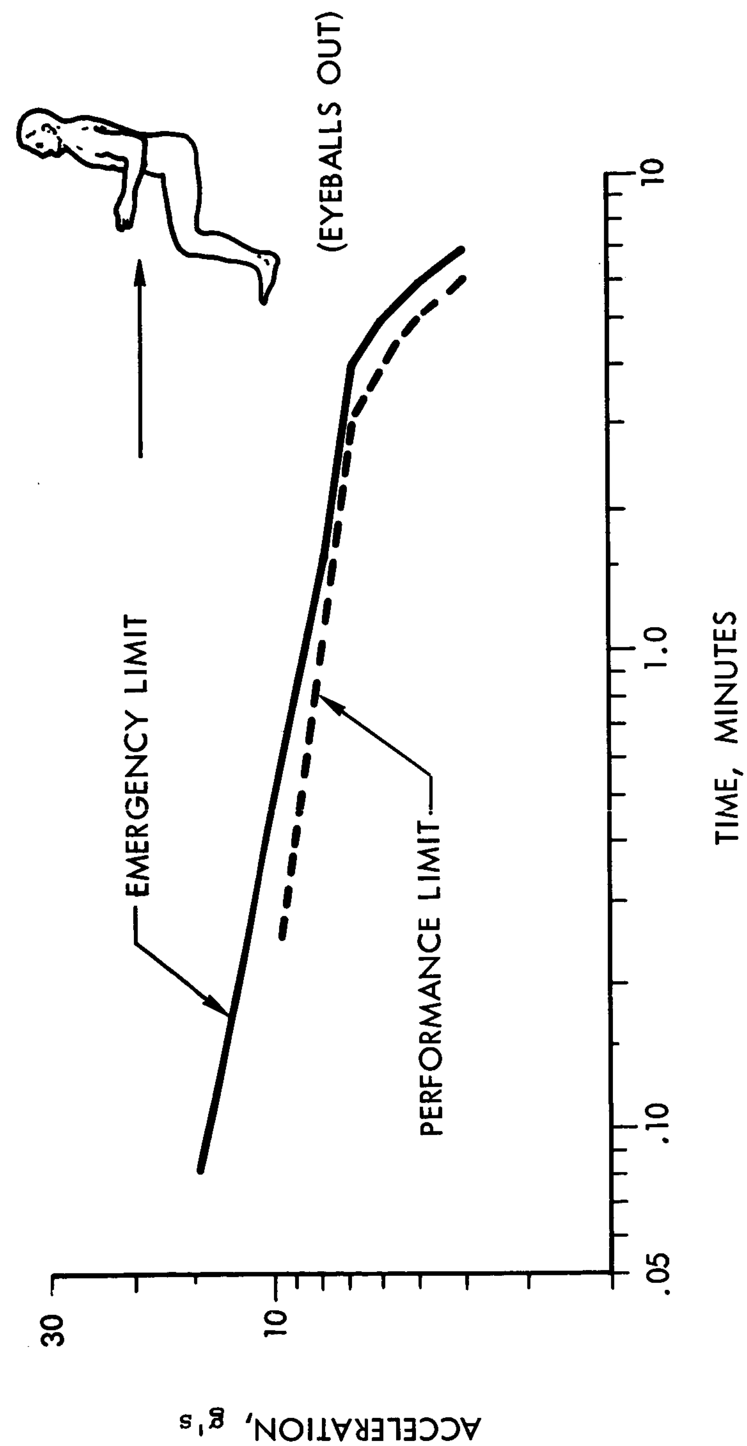

Figure 57. Sustained Acceleration - Eyeballs Out

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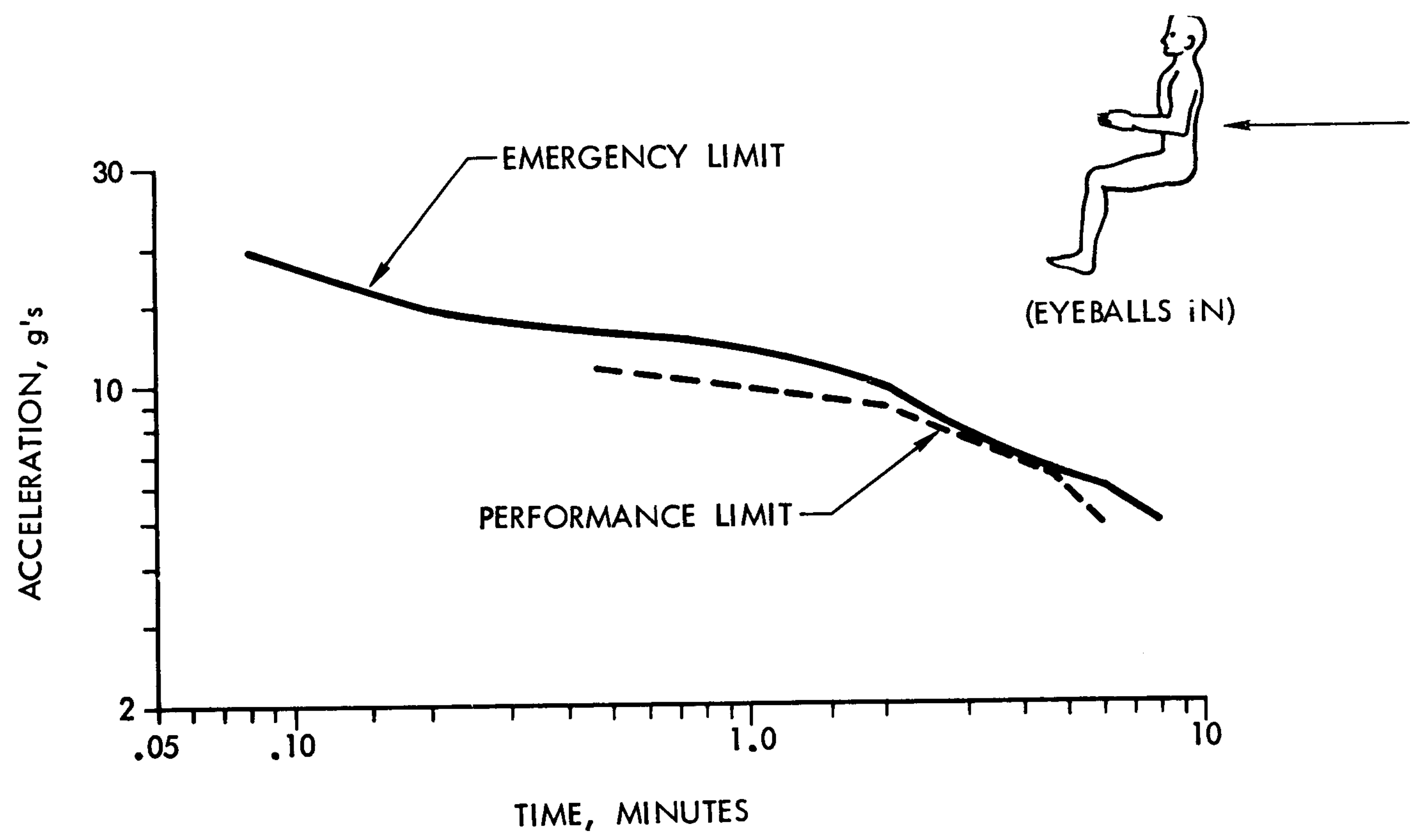

Figure 58. Sustained Acceleration - Eyeballs In

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CON[redacted]

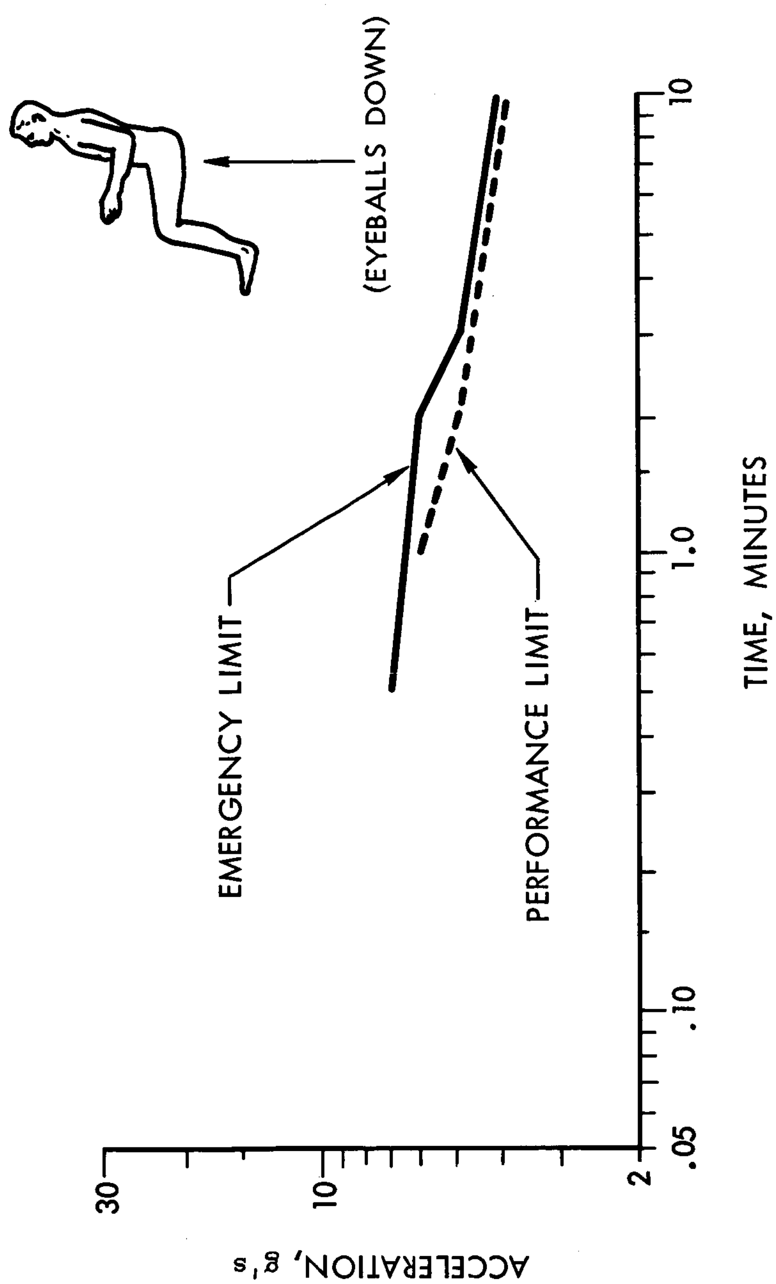

Figure 59. Sustained Acceleration - Eyeballs Down

[redacted]

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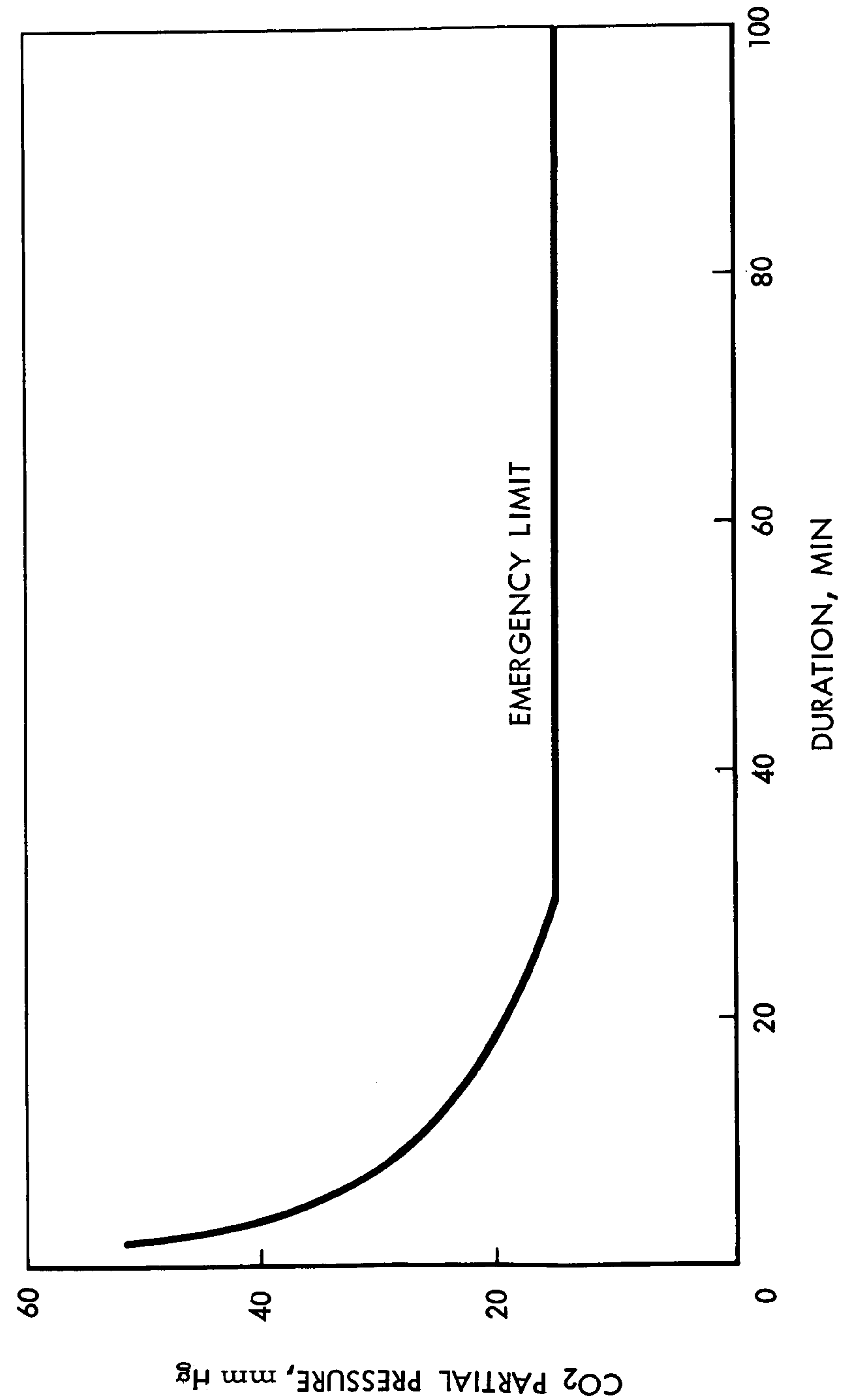

Figure 60. Emergency Carbon Dioxide Limit

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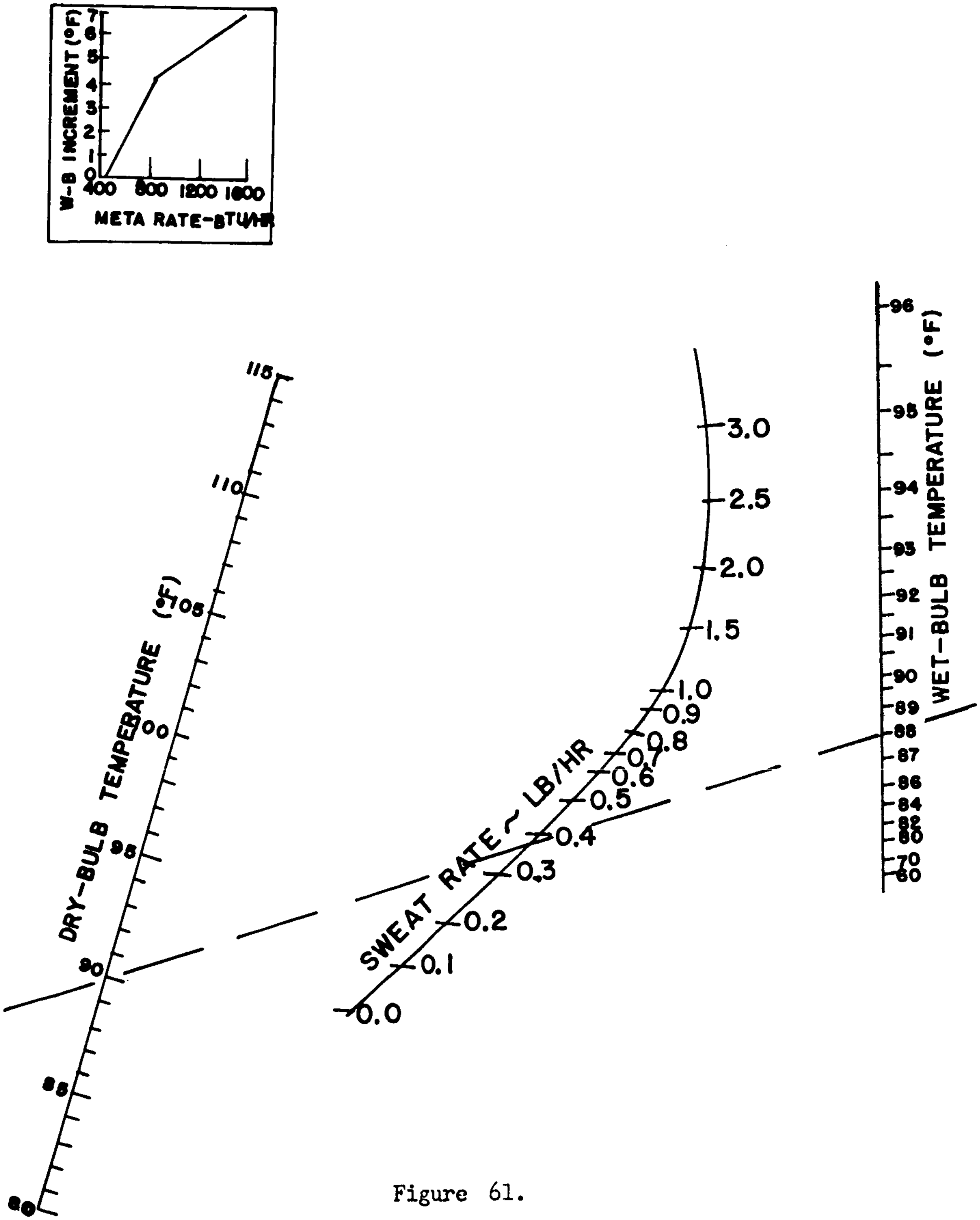

Figure 61.

NOMOGRAM FOR THE CALCULATION OF SWEAT RATE

(THE INSET CHART GIVES THE INCREMENT TO BE ADDED TO THE WET-BULB TEMPERATURE FOR METABOLIC RATES BETWEEN 400 AND 1600 BTU/HR.)

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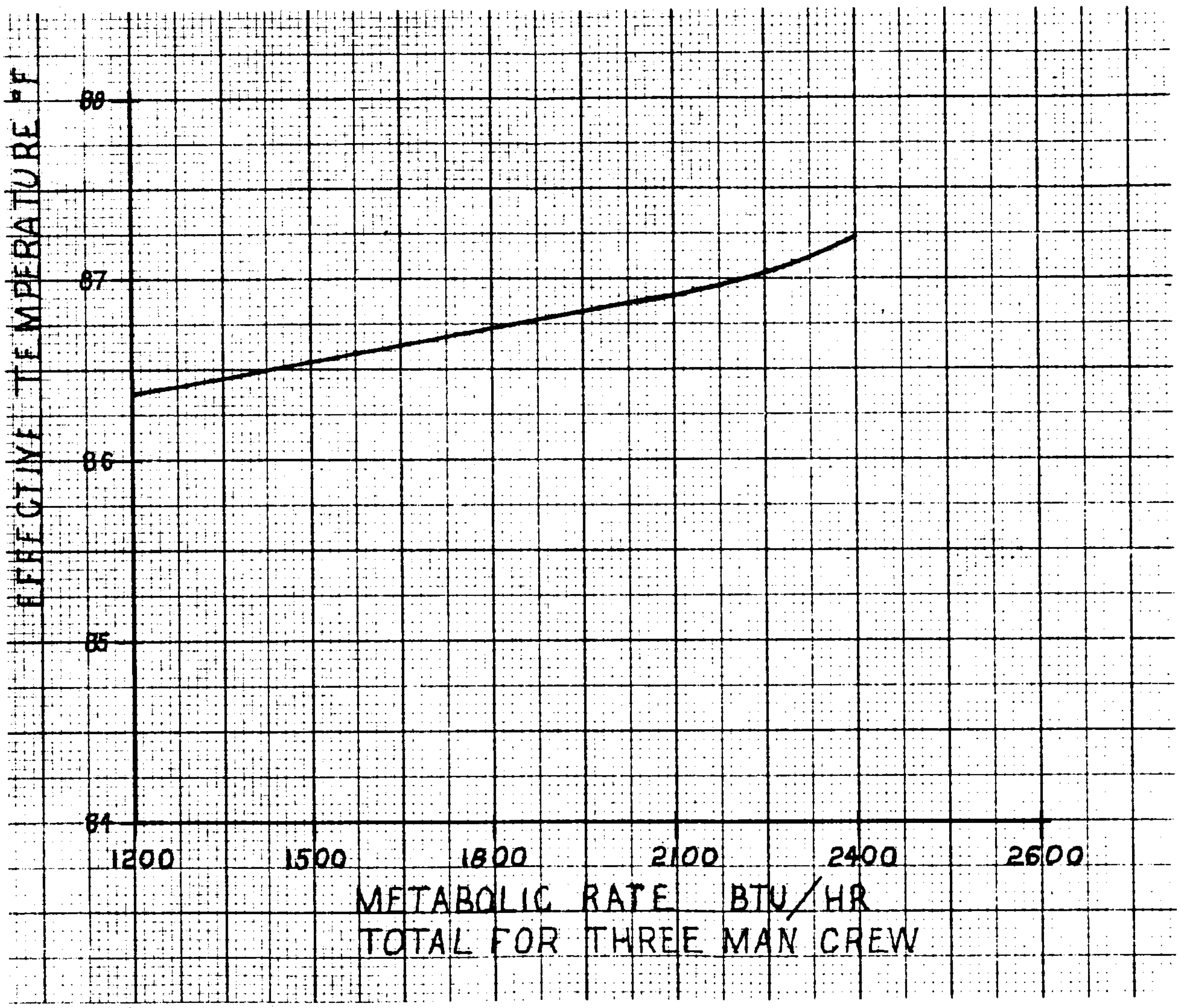

Figure 62. Post Landing ECS, Allowable
Effective Temperatures

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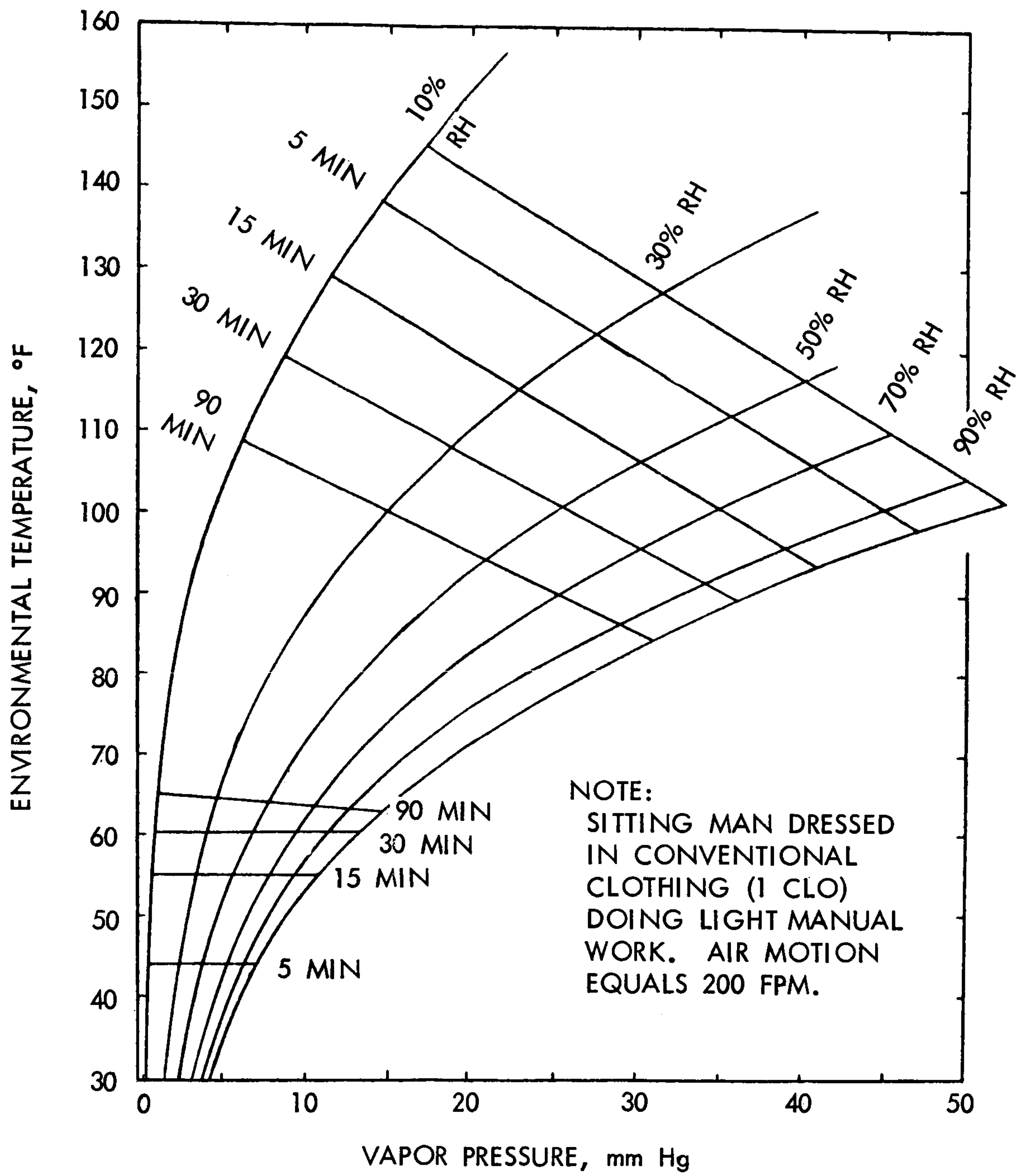

Figure 63. Temperature and Humidity Nominal Limit

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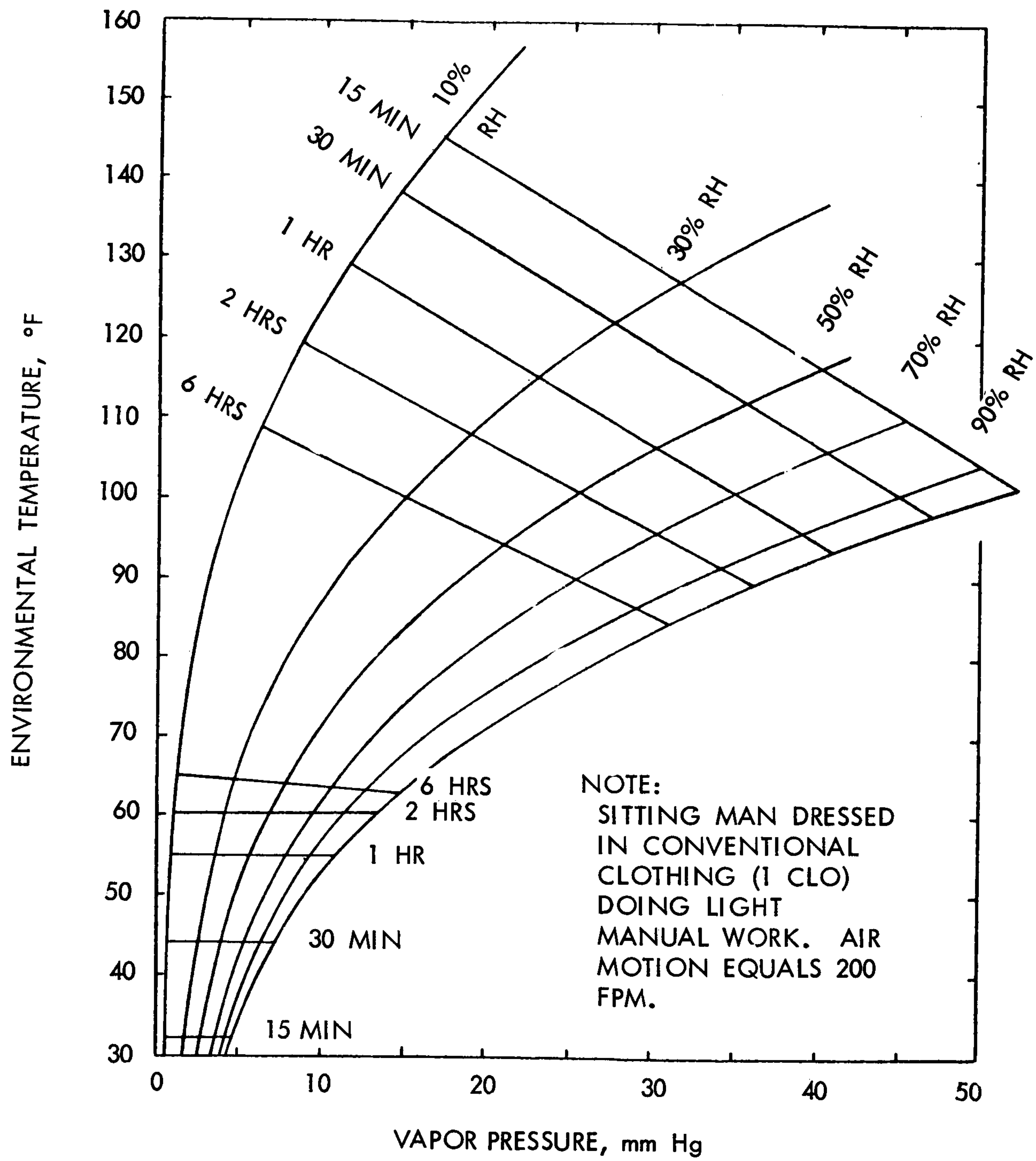
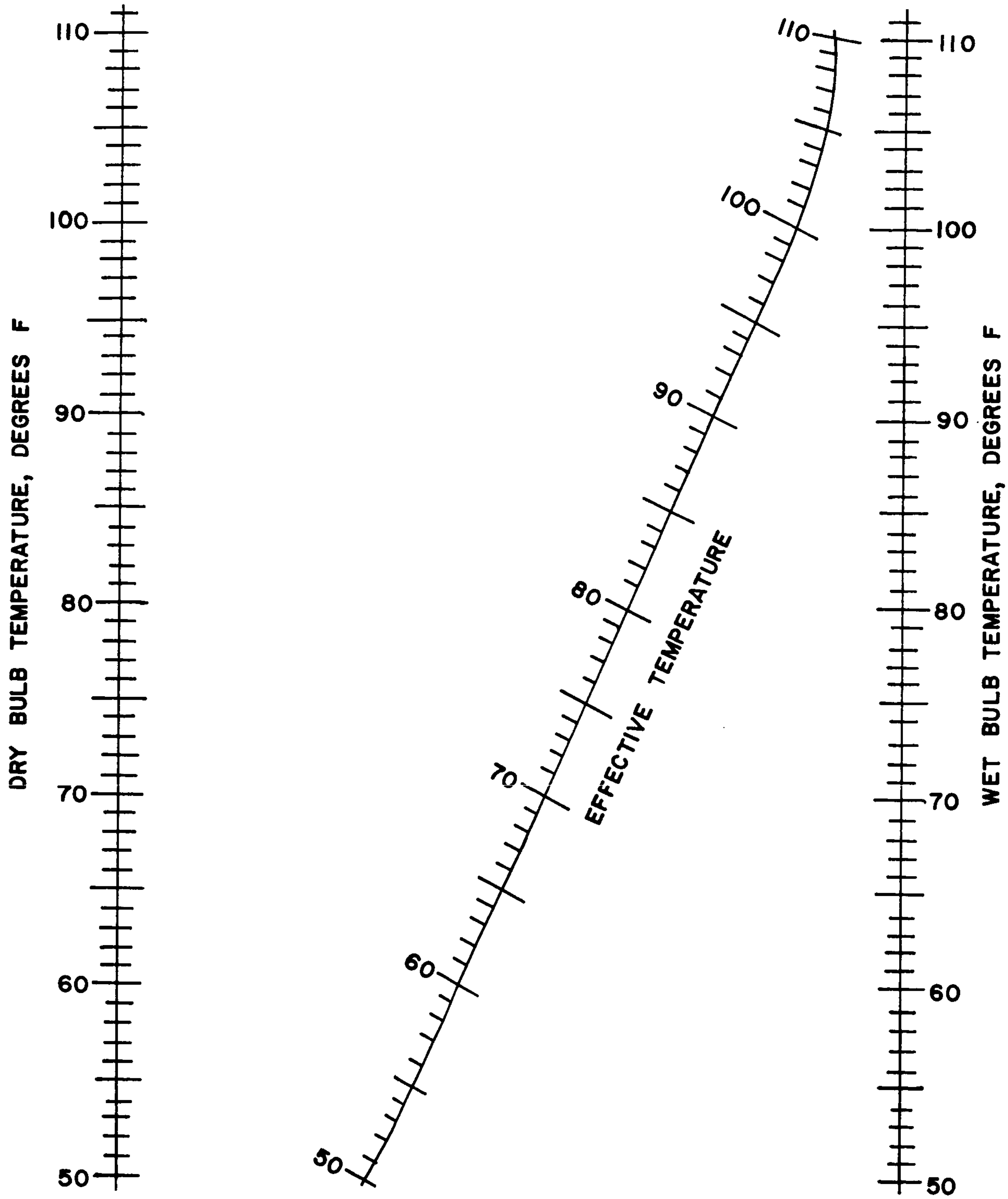


Figure 64 Temperature and Humidity Emergency Limit

~~CONFIDENTIAL~~



NOMOGRAM FOR CALCULATION OF EFFECTIVE AIR TEMPERATURE

(APPLICABLE TO PERSONS AT REST, STRIPPED TO THE WAIST IN STILL AIR)

Figure 65.

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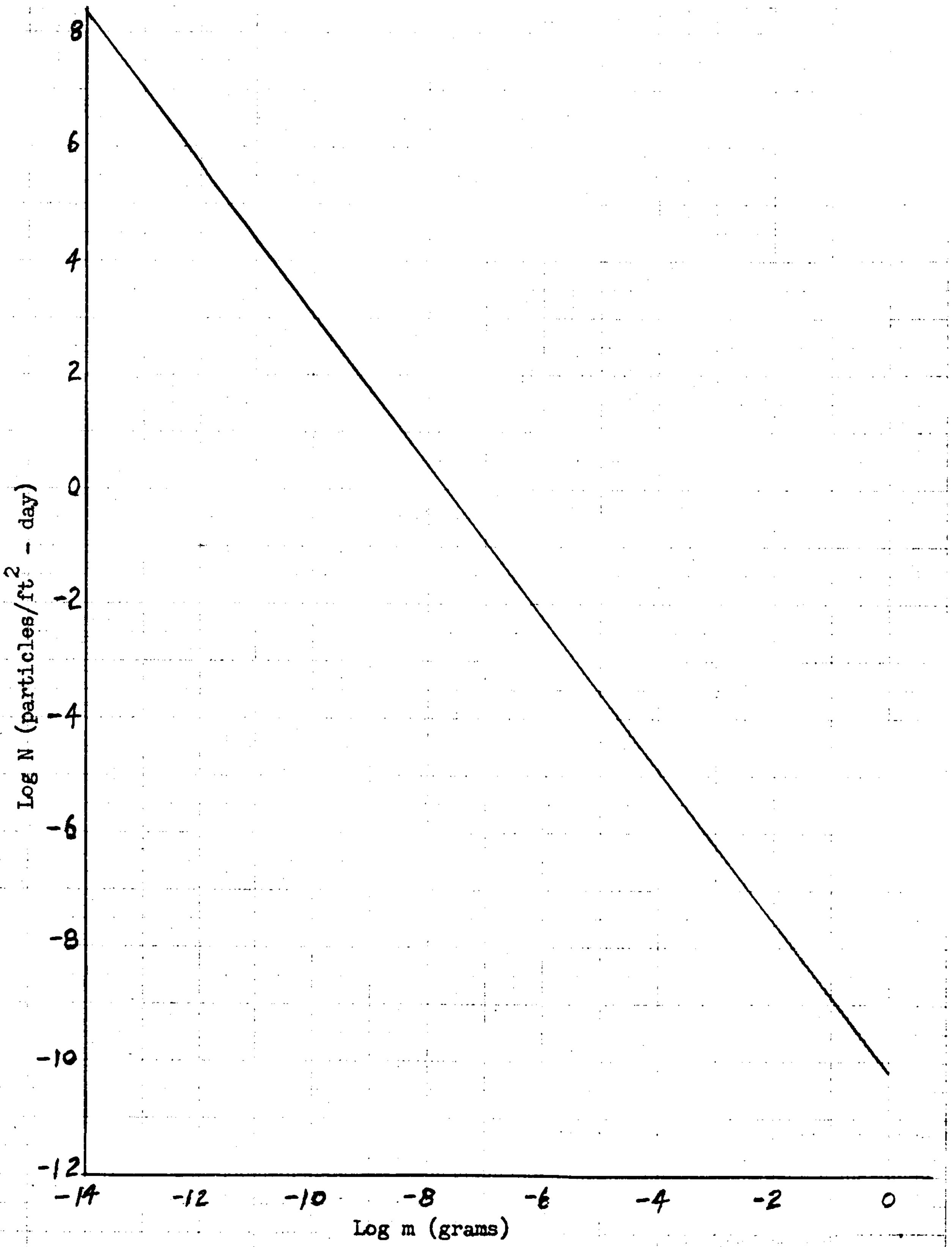

Figure 66. Sporadic Near-Earth, Cislunar, and near-lunar meteroid environment

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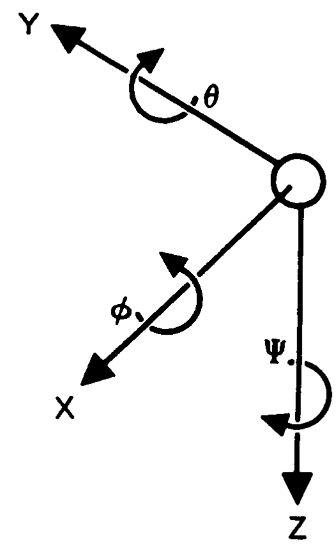
Table 1. Reference Axes

Positive direction of axes and angles (forces and moments) are shown by arrows. (When launch vehicle is at a launch angle of 90°, the positive "X" direction is vertically upwards.)



Axis		Moment About Axis		
Designation	Symbol	Designation	Symbol	Positive Direction
Longitudinal	X	Rolling	L	Y ⟶ Z
Lateral	Y	Pitching	M	Z ⟶ X
Normal	Z	Yawing	N	X ⟶ Y

Force	Angle		Velocities	
(Parallel to Axis Symbol)	Designation	Symbol	Linear (Components along Axis)	Angular
X	Roll	ϕ	U	p
Y	Pitch	θ	V	q
Z	Yaw	ψ	W	r

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[redacted]

Table 2. Estimates of Metabolic Rate, Thermal Balance and Water

General Requirements for Apollo Crew Member

Per Man		Command Module Routine Flight Per Day	Command Module Emergency Decompression Per Day
Heat Output	Btu	11, 200	12, 000
Oxygen	lb	1.84	1.97
Carbon Dioxide	lb	2.12	2.27
Latent Heat (lungs)	Btu	2, 800	3, 000
Latent Heat (sweat)	Btu	1, 310	7, 230
Sensible Heat	Btu	7, 090	1, 870
Urinary Loss	g	1, 200	1, 200
Sweat Loss	g	600	3, 140
Lung Loss	g	1, 200	1, 300
Total Water Requirement	g	3, 000	5, 640
Total Water Requirement	lb	6.6	12.4

[redacted]